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ROBERTSON YOUNGQUIST, Editor

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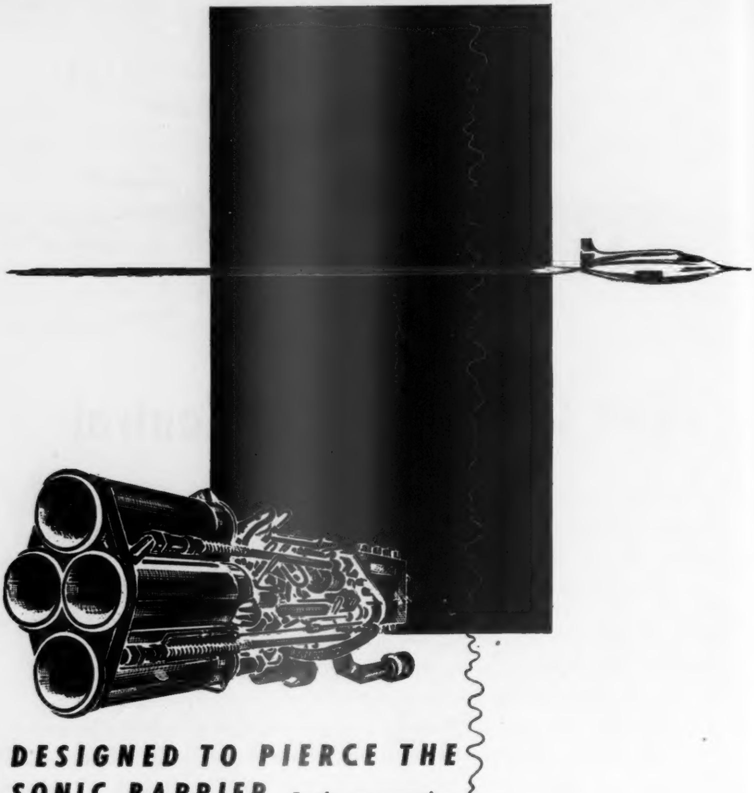

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CREATION OF HIGH-PRESSURE GAS SOURCE FOR ROCKET-MOTOR-PROPELLANT SUPPLY SYSTEMS

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CHAMBER pressures of 300 psi are common in today's rocket motors and all indications are that higher pressures will be used in the future. To force propellants into the chambers against these pressures, feed pressures of 350 to 500 psi are required. Current practice is to obtain the requisite propellant pressure by means of turbine-driven pumps or by pressurizing the propellant tanks with gas. The former system is most applicable to high-thrust long-duration liquid-propellant motors, that is, in airplanes and large missiles. Even in such applications, however, there is a place for gas-pressurized propellant supply because the turbine must be fed with its propellant. In some cases pressurization of incoming propellants is required to prevent cavitation at pump inlets.

Until safer more reliable methods are perfected for this purpose gas pressurization must be derived from storage of inert gases in the vehicle affected. Obviously space considerations dictate that the gas be stored at high pressure, with subsequent reduction to the working pressures required. In addition, the problem of obtaining the high-pressure gas required is complicated by the fact that it must be uncontaminated by hydrocarbons, because it will eventually come into intimate contact with powerful oxidizers. It must also be free from moisture because the temperatures inherent in reduction of pressure would freeze any moisture in mechanisms downstream of the pressure reducer.

Since the inception of its work in the rocket field some four years ago, the authors' company has been using pressures in the order of 4500 to 5500 psi for gas storage in air-borne vehicles. It was found early in the program that the purity considerations mentioned above made it impossible to obtain such high-pressure gas commercially, because the compressors and "water-pumping" systems used contaminate the gas. Use

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of the gas in aircraft and missiles, moreover, makes other features desirable in the high-pressure creating devices. For example, they should be portable in order to facilitate flight testing from various sites; they should have relatively high rates of pressurization; and, most important of all, they should be foolproof. These features were not obtainable in commercial machines either, so it became necessary to develop specialized units. This paper describes two of the resultant systems: liquid-gas evaporators and dry-gas pressure boosters.

Liquefied-Gas Evaporator

The liquefied-gas evaporator has the following advantages: (*a*) The total elimination of a gas-pump system, thus offering operation free from two problems: first, that a pump system induces oil into the pressurizing gas causing pollution, and second, that pump sealing and packing difficulties are extreme at pressures of 4000–5000 psi; (*b*) the elimination of a large number of gas-storage cylinders; and (*c*) the provision of a portable source of supply.

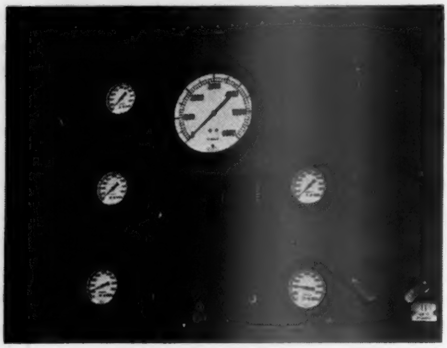


FIG. 1 PANEL OF LIQUEFIED-GAS EVAPORATED UNIT

An examination of the schematic diagram, Fig. 1, of such a system shows the operating cycle to be exceedingly simple. A high-pressure sphere is filled with liquid nitrogen from a commercial service cart by means of a flexible hose. The sphere vent is then closed and the liquid service valve is cracked, allowing liquid to flow to the bottom of the heat exchanger. Heat is transferred from an electrically heated air supply to the liquid, causing the liquid in the exchanger tubes to boil off and leave the exchanger as a gas at a temperature of approximately 0 F. The gas then flows to a manifold from which it can be distributed as desired. A small dry-gas pump, enclosed in a high-pressure vessel, is used as a motivating force for continuing the flow of liquid through the heat exchanger. This pump slightly boosts the pressure of a small amount of gas tapped from the manifold and discharges it to the top of the liquid sphere.

The first such evaporator plant was developed specifically for the pressurization requirements of the Bell XS-1. Based on the success of this large unit a smaller evaporator was designed for missile service where less storage volume is required. This unit has been built into a two-wheeled

trailer approximately $6\,^{3}/_{4}$ ft $\times$ 4 ft $\times$ $3\,^{3}/_{4}$ ft. All pressure gages, temperature indicators, valves, and switches required for operation are mounted on a panel at the rear of the trailer. The other components (liquid-gas sphere, heat exchanger, boost pump, and gas containers) are located within a sheet-metal cover. The approximate rate of evaporation is one gal per min which will produce 87.5 cu ft of pollution-free gases at standard temperature and pressure. This allows a storage sphere of 1.5 cu ft to be charged to 5000 psi in the short time of six minutes. Built into the evaporator are four such storage containers.

Dry-Gas Pressure Booster

Although nitrogen gas has been used extensively, by means of the liquefied-gas evaporator in rocket power-plant fuel-feed systems, it has some disadvantages as a propellant-tank pressurizing agent, especially when liquid oxygen is one of the propellants. As nitrogen gas impinges against liquid oxygen, it is dissolved into solution; consequently there is no pressure rise until a sufficient nitrogen-gas insulation blanket has been created. As a result, there is a considerable time delay, in some cases as long as 30 sec, in the pressurization of the propellant tanks. Also as a result of the solubility of nitrogen gas in liquid oxygen, the gas factor has been as high as 2.9 and 3. This gas factor is the ratio of the volume of gas

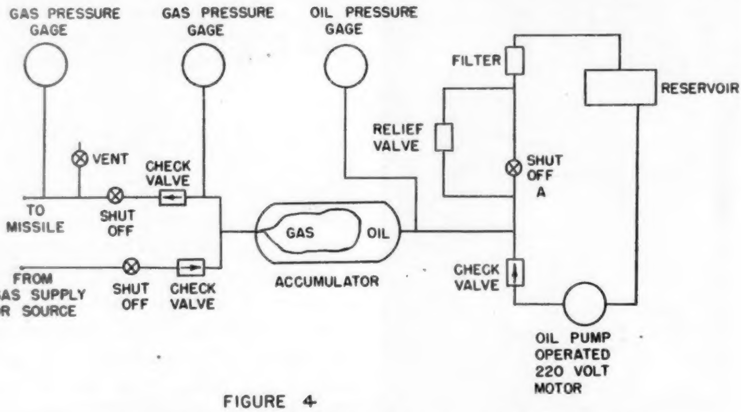


FIG. 2 SCHEMATIC DIAGRAM OF DRY-GAS PRESSURE-BOOSTER UNIT

at propellant-tank pressure required to displace the propellants to the volume of the displaced propellants. Helium, however, gives instant pressurization of liquid-oxygen propellant tanks and a gas factor of 1.8 to 1.9.

Since the liquefaction of helium is as yet in the experimental stage, the previously developed evaporator could not be used. Oil-free gaseous helium, commercially available at pressures up to 2200 psi, could be used

if the previously mentioned pumping difficulties could be overcome. Commercial compressors could boost the helium-gas pressure from a minimum of 500 psi to as high as 5000 psi, but are expensive and could not be adapted for use as portable field units. They would be subject also to the purity limitations mentioned previously. It was necessary, therefore, to develop a specialized unit. The result is an oil-free dry-gas helium-pressure booster (Model 05) which has been in satisfactory use since December, 1947. This booster, shown schematically in Fig. 2, is advantageous in the following respects: (*a*) It is portable for field service use; (*b*) commercially standard valves, accumulators, and tubing are used; and (*c*) it requires a minimum amount of maintenance.

Operation, which requires the attention of an operator at all times, is as follows:

1 Gas is delivered from a storage source into the accumulator's bladder, thereby displacing all oil into the reservoir. This bladder is used to prevent oil contamination of the helium gas.

2 When all gages have stabilized, the "shutoff" valve "A" is closed.

3 The pump is started, filling the accumulator with oil and thus compressing the helium gas in the bladder. The pump is operated until the accumulator is filled to 90 per cent of the total capacity. Caution must be taken, as it is possible to extrude the bladder through the air-discharge end of the accumulator.

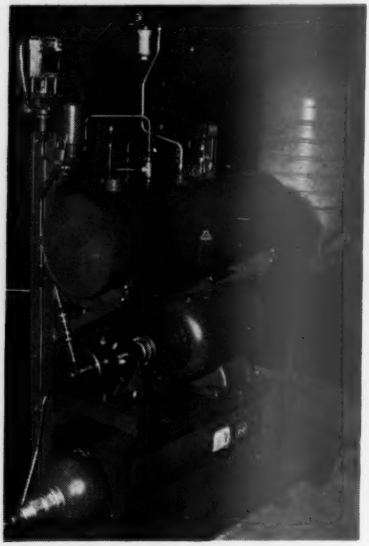


FIG. 3 VIEW OF ELEMENTS OF MODEL 05B DRY-GAS PRESSURE-BOOSTER UNIT

A second unit, Fig. 3, which is fully automatic, has also been developed and is known as the Model 05B. This unit makes use of a larger pump so that the total time to charge a 1-cu-ft sphere (see Fig. 4) to a final pressure of 4000 to 5000 psi, with various available source pressures is decreased. The operation of the second unit differs from that of the first only in the use of a 5000-psi pressure switch in conjunction with an electric timer to control the pressurizing and exhausting cycle.

At the present time a third and much larger unit, designated the Model 06, is being constructed. This unit, also completely automatic, makes use of two V-8 internal-combustion engines to operate the hydraulic pumps. Its operation is very rapid, since cycling between accumulators is em-

ployed rather than shutting off the pumps while oil is being expelled from an accumulator. In the Model 06, the time to charge a 10-cu-ft storage sphere with various available source pressures is approximately $^4/_{10}$ of the time regained by the Model 05B to charge a 1-cu-ft storage sphere.

A system of such complexity obviously has potential operational difficulties. Since the rate of gas compression in a booster of this size is extremely high, the gas leaves the compressor at a fairly high temperature and the incorporation of a cooling system to remove this heat of compression may be required. Also, due to this high rate of compression, the gas leaves the booster at a high velocity. Any contaminants resulting from bladder failure must be detected and the system shut down immediately before the contaminants reach the storage bottles. These possibilities have been foreseen and appropriate devices have been incorporated in the system to monitor its operation.

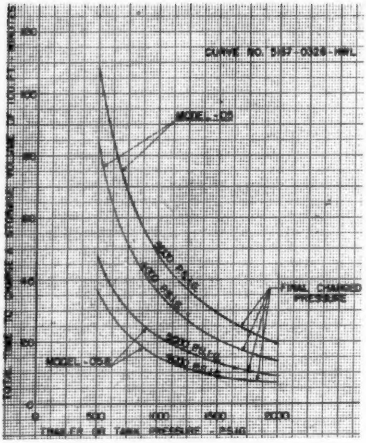


FIG. 4 TIME TO CHARGE A 1-CU-FT VESSEL AS A FUNCTION OF SOURCE PRESSURE, FOR MODEL 05 AND 05B UNITS

Conclusions

The liquid-gas evaporators and the dry-gas pressure boosters described are characteristic solutions to the problems existing. The present trend, however, toward storage pressures in an even higher range magnifies the problems already entailed in procuring equipment from commercial sources. The obvious course of further effort lies in the development of specialized accumulators, automatic valving, and pressure switches capable of reliable operation under these extreme conditions.

THE PROSPECTS OF JET-REACTION FLIGHT—PART III[1]

By Eugen Saenger

Consulting Engineer, Paris, France

In preceding supplements, Dr. Saenger characterized jet-reaction flight as a "new branch of aeronautical engineering." He reviewed the introduction of the resonance duct and discussed theoretical aspects of the ramjet and rocket power plants. In Part II he commenced a discussion of aircraft design as it is affected by aerodynamics of supersonic flight at great altitudes. In Part III he continues the subject of the design of reaction aircraft and comments on the application of atomic power to reaction flight.

Subsonic Ramjet Aircraft

THE relatively low efficiency of ramjet engines in the subsonic domain indicates atomic heating for subsonic ramjet aircraft if the latter have to be used for transport purposes over long distances or, if chemical heating is foreseen, that they be used for special purposes, for which medium flight duration and range suffice, for example, for military or scientific missions. In the case of fighters, where the ramjet couples the high climbing speed of the rocket fighter with the flight duration of the turbojet fighter, the low construction cost and the possibility of utilizing cruder types of fuel speak in favor of the ramjet.

A subsonic aircraft powered with a pure ramjet engine is presented schematically in Fig. 17.

The duct of 2400-mm maximum diameter and 11,500-mm total length has an external diffuser and a fixed, nonadjustable efflux cross section F_4, and forms the fuselage of the aircraft.

The upper part carries a cockpit, cabins for the crew, fuel tanks, a hold for the useful load, and a tail assembly with rudder and elevators. The mid-set wing passes through the free space of the diffuser.

The landing skid retracts into the thick part of the external diffuser and the take-off can be effected by releasing the aircraft from a large carrier aircraft at altitudes by catapulting it, or with the aid of a jettisonable undercarriage and take-off rockets.

The double-walled monocoque construction of the diffuser forms the backbone of the aircraft and allows for the assembly of all the component

[1] Reprinted from *Inter Avia*, vol. III, November, 1948, pp. 617–622. This concludes Dr. Saenger's paper, parts I and II of which were reprinted in the March and June, 1949, issues of ARS JOURNAL.

parts mentioned. The diffuser also contains the essential elements of the power plant; the injection grill, the fuel supply system, the armatures or, as the case may be, the combustion grill for coal or the lattice of atomic heating.

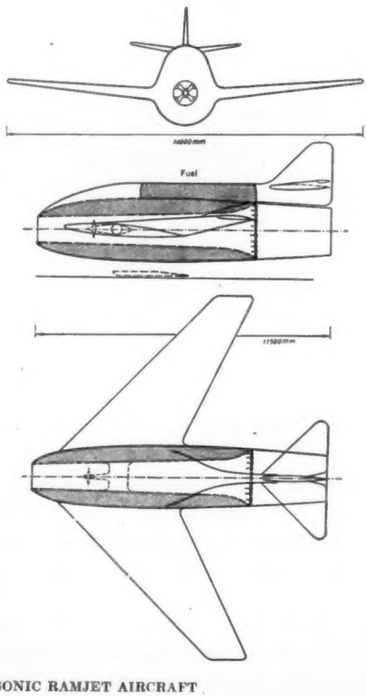
On the other hand, the combustion chamber and the flame nozzle, in view of their high working temperature, are located at a distance from the rest of the aircraft and merely consist of a simple steel coque which, because of its rapid detrition by the burners in the zone of the injection grill, can be dismantled and replaced.

A special mechanism for regulating the cross section would be superfluous in view of the automatic regulating action of the external diffuser which, with $X/l = 0.7$, can be used for Mach numbers approaching 0.9. Owing to the pronounced sweepback of the wing, the polar divergence of the airframe does not become noticeable until this Mach number has been exceeded.

The aircraft weighs 4000 kg empty, has an initial gross weight of 20,000 kg, and none of its essential parts move. The aircraft represents a pure sheet-metal structure, if one makes exception of the few auxiliary devices for fuel injection, cabin ventilation, landing skid retraction, and flap adjustment, etc.

The take-off can be effected with the aid of ordinary assisting rockets attached on either side of the fuselage.

Supersonic Ramjet Aircraft: A sketch of an experimental supersonic aircraft powered with a ramjet engine is shown in Fig. 18.



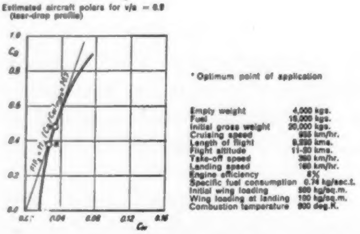


FIG. 17 LONG-RANGE SUBSONIC RAMJET AIRCRAFT

The speed range should extend up to Mach 3; the basic configuration corresponds to the subsonic aircraft mentioned earlier.

The demands as regards flow at supersonic speeds are satisfied by the choice of a pure internal diffuser for the engine, the thin, rhombic airfoil section of the wing and tail surfaces, the far rearward position of the main

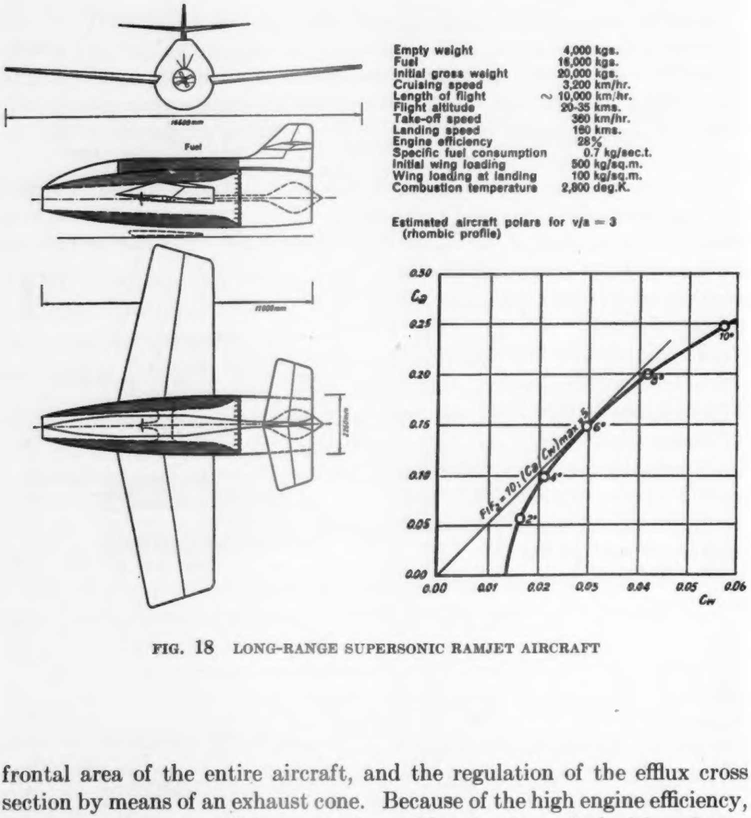


FIG. 18 LONG-RANGE SUPERSONIC RAMJET AIRCRAFT

frontal area of the entire aircraft, and the regulation of the efflux cross section by means of an exhaust cone. Because of the high engine efficiency, the maximum range will be great when oil heating is applied, although one must expect the airframe to have an unfavorable lift-drag ratio at a speed of Mach 3.

Long-Range Rocket Aircraft: Fig. 19 shows the external aspects of an ultra-high-speed supersonic rocket aircraft, taking into account the demands of gas dynamics at high Mach numbers.

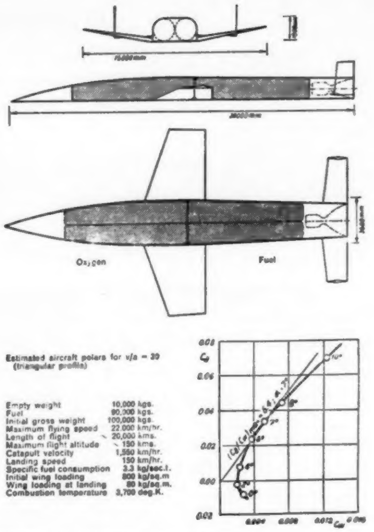


FIG. 19 LONG-RANGE ROCKET AIRCRAFT

The fuselage nose takes the form of an extremely slender ogive cut along its longitudinal axis in such a way that a plane surface forms the underside of the fuselage. At the level of the wing, the semiogive transforms progressively into a spacious box of rectangular section, and toward the rear it tapers at approximately the same rate as the central cross section of the fuselage. The large dimensions of the rear wall terminating the fuselage are dictated by those of the efflux cross section of the rocket engine.

The relatively small stub wings are mainly for stabilization purposes during flight and landing; their airfoil section is the well-known triangular wedge-type profile, of which the maximum thickness of $^1/_{20}$ of the chord is located at $^2/_3$ of the chord. There is no necessity for an angle of incidence between the wing and fuselage with the result that, in view of the low-wing arrangement chosen, the supporting surfaces of the fuselage and wing continue into each other without disturbance. The configuration of the tail assembly is independent of the airstream created in the ambient medium by the rocket efflux, since the rocket engine will never have to function at subsonic speeds.

It is foreseen that the take-off should be 100 tons and the empty weight 10 tons. The dimensions of the wings are determined by the fact that during catapult launches at Mach 1.5, approximately 38 per cent of the weight is made dependent on the supporting properties of the fuselage and 62 per cent on the wing. The participation of the fuselage increases very sharply at high Mach numbers, and can attain 66 per cent, later to fall to about 40 per cent, during landings at subsonic speed.

The configuration, at present seemingly odd, of such aircraft designed in accordance with the laws of supersonic flow, is justified by the fact that long-distance flight will be pure motorless flight, using only the kinetic energy; indeed a rocket aircraft with a lift-drag ratio of 6.4 already utilizes 99 per cent of its kinetic energy while passing from Mach 30 to Mach 3, thus within the speed range of Newton, and only 1 per cent at lower Mach numbers at which other aircraft configurations will lead to more favorable values of lift-drag ratio.

Flight Performance

Thrust versus Mach Number: The flight performance depends chiefly on the variations in engine or propeller thrust versus the flying speed—a function which is presented for a number of large engines in Fig. 20. Noticeable are:

1 The high static thrust of propeller power plants. The thrust diminishes rapidly as the flying speed augments and, within a wide speed range, this thrust is inversely proportional to the flying speed and proportional to the output (constant) of the engine, in other words, PL/v.

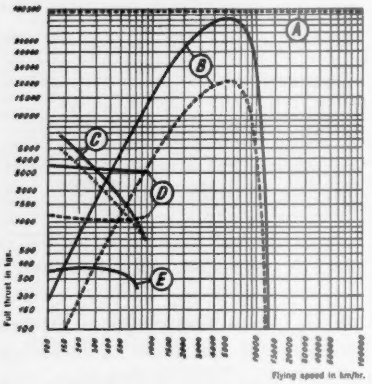


FIG. 20 FULL THRUST OF LARGE ENGINES AT SEA LEVEL (FULL LINE) AND AT 12,000 METERS (DOTTED LINE)

(A—100-ton rocket; B—2500-mm-diam ramjet; C—4000-hp piston engine with propeller; D—axial-flow turbine of 1250 mm diam; E—resonance duct of 560 mm diam.)

2 The thrust of turbojets and rocket engines is constant. It will also be noticed that the thrust of rocket engines does not depend on the altitude.

3 The ramjet engine has no static thrust, and its thrust increases initially as the square of the flying speed, then more slowly, ultimately to reach a high maximum. It will also be noticed that the resonance duct has a feeble thrust.

Application Points on the Polar Curves: It is known that, in the case of propeller aircraft, one obtains maximum values of range and flight duration when the point of application coincides with the point $(c_w/c_a)_{\min}$ of the aircraft polars, thus at the optimum lift-drag ratio. Similar conclusions can be reached for rocket aircraft.

As long as the polars are independent of the speed, one obtains in steady horizontal flight for example, the minimum consumption of fuel per kg gross weight and meter distance or second flight duration, under the conditions given.

For pure rocket aircraft and, with a certain degree of approximation, for turbojet aircraft, the consumption per kg gross weight, and meter distance is at a minimum when the speed is the highest possible value and (c_w/c_a) at a minimum; this is attained by seeking the optimum altitude or suitably choosing the wing dimensions. If the aircraft and flying altitude are prescribed, the speed should be chosen in such a way that (c_w^2/c_a) is at a minimum.

For these two classes of aircraft, the consumption per kg gross weight and second flight duration is at a minimum when (c_w/c_a) is also at a minimum.

For ramjet aircraft, the consumption per kg gross weight and meter distance will be at a minimum when the speed is as high as possible and $(c_w/c_a)\ (c_aF/4F_1 + 1)$ minimum. If the wing characteristics and the flight altitude are predetermined, the speed must be chosen in such a way that $[c_w/\sqrt{c_a}\ (c_wF/4F_1 + 1)]$ is at a minimum.

If the polars are dependent on the flying speed, as they are in the sonic and supersonic ranges, such simple relations can no longer be used.

The conditions also change in unsteady flight; for instance, it is evident that the motorless supersonic flight of pure rocket aircraft will be best when (c_w/c_a) is at a minimum.

Flight Performance of Ramjet Aircraft. The range s of the subsonic jet aircraft chemically heated, in horizontal flight, is given by the following relation:

$$s = \frac{(v/a)^2 \ln G_0G}{2gc_p/H_u \cdot c_w/c_a \cdot (c_wF/4F_1 + 1) \cdot [1 + (v/a)^2 \kappa g R/2000]/\kappa g R}$$

whereby v/a is the Mach number of flight, G_0 the initial weight and G the end weight in horizontal flight (thus $\Delta G = G_0 = G$ is the total fuel consumption), g acceleration of gravity, c_p the specific heat of the combustion gases at constant pressure, F the wing area, F_1 the cross-sectional area of the air inlet into the cylinder, R the individual gas constant, κ its adiabatic exponent, and F_2 the frontal area of the duct.

If one accepts the relation $c_{w1} = a + (bc_a)^2$ for the aircraft polars and $c_{w2} = cF_2/F$ for the drag of the duct, one can determine an optimum ratio of the wing area F to the frontal area of the duct F_2 under the condition that the fuel consumption per meter distance and kg gross weight

$$\frac{dG}{Gds} = \frac{2gc_p}{H_u} \cdot \frac{c_w}{c_a} \left(\frac{c_wF}{4F_1} + 1\right) \frac{1 + \left(\frac{v}{a}\right)^2 \kappa \cdot gR/2000}{\left(\frac{v}{a}\right)^2 \kappa \cdot gR}$$

is at a minimum.

For example, one obtains for $a = 0.01$; $b = 0.07$; $c = 0.07$; and $F_1/F_2 = 0.16$, an optimum value $F/F_2 = 11$.

If one uses the polar curve of the now determined aircraft at its most favorable point ($c_a = 0.382$ according to the above figures), one will fulfill all the conditions prerequisite to optimum range.

For a Mach number of $v/a = 0.9$ (which seems reasonable with the polar curve if the wing is sharply swept back) and a weight of fuel amounting to 60 or 80 per cent of the take-off weight, G_0, the above formulas give a range of 5280 or 9150 km, thus interesting values which seem to show that the subsonic ramjet aircraft with oil heating could be used for transportation purposes, as an unmanned transport, or as long-range fighter.

It should furthermore be stressed that this result is independent of the flight altitude and the wing loading, that is, it can be achieved at any flight altitude if a judicious selection of wing loading is made. Nevertheless, the wing loadings of below 500 kg/sq m which occur in practice, indicate flight altitudes of over 10,000 meters, which continue to increase during horizontal flight as a result of the reduction in wing loading.

One can present corresponding considerations for the ascent of chemically heated subsonic ramjet aircraft, thus for the minimum consumption per meter altitude gained and kg gross weight, and T. Guillot obtains for $F/F_2 = 0$ the most economic theoretical ascent, this being a vertical ascent. The practical consequences to be drawn are that aircraft which have above all to climb must feature a small ratio of F/F_2; those which must simultaneously have good climbing and horizontal-flight characteristics (fighters) a value of F/F_2 of about 6; and primarily long-range aircraft a high value of F/F_2 of the order of 11, so that climbing and horizontal flight together always work out at an optimum. Finally, one can also vary the ratio F_1/F_2, which we have hitherto regarded as constant, on the basis of experience gained in conjunction with engines; and in this way one meets with conditions that are a little more favorable. In conclusion, it must be noted that the range formula reveals the favorable influence of high wing-aspect ratio, and that the optimum combustion temperatures for climbing and horizontal flight are moderate.

The study of supersonic flight is a little more complicated and, in view of the high engine efficiency and despite the less favorable lift-drag ratio, indicates ranges of the order of 10,000 km in pure horizontal flight, if this study is carried out with care.

Finally, as has been stated earlier, the utilization of atomic energy in the subsonic and supersonic domains results at all flight altitudes in practically unlimited range and flight duration.

Flight Performance of Long-Range Rocket Aircraft: There are two fundamental flight procedures available to this category of aircraft: acceleration up to the moment the flying speed is equal to the efflux velocity, followed by flight at constant speed and reduced thrust; or acceleration up to a speed such that the subsequent nonpropelled gliding flight enables the required range to be attained. The latter procedure—and it is in this that it distinguishes itself absolutely from the former—leads to greater range and simpler rocket engines, the consumption remaining the same. We shall restrict our consideration to this procedure.

The introduction to the section on rocket engines[2] of this article gives an idea of the ranges which may be realized by this procedure and also shows that, by dividing the rocket into several stages, one can obtain initial gliding speeds:

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[2] See ARS JOURNAL, March, 1949, p. 35.

$$v_{1_{st}} = k_1c_1 \cdot \ln \frac{G_0}{G_1 + \Delta G_1} + k_2c_2 \cdot \ln \frac{G_1}{G_2 + \Delta G_2} + \ldots k_nc_n \ln \frac{G_{(n-1)}}{G_n}$$

(ΔG is the empty weight of the carrier aircraft) which are much higher than those attainable by "single-go" propulsion $v_1 = kc \ln G_0/G$, and that a large part of the dead weight does not need to be accelerated up to v_1, but is abandoned beforehand, with the result that $G_n < G$, and the range increases as a result of the multistage propulsion. For example, the initial gliding speed of a two-stage rocket aircraft is double that of a single-stage type, if $k_1 = k_2 = k$; $G_0/(G_1 + \Delta G_2) = G_1/(G_2 + \Delta G_2) = G_0/G$; $c_1 = c_2 = c$, from which it may be deduced that the ranges, according to the section on rocket engines, are increased at least fourfold.

In the case of multistage rocket propulsion (supersonic catapult, carrier rockets as primary stages) the v_1 values at each stage simply add together; then the values of k must be ranged in suitable gradation and made to tend strongly toward unity in the higher stages. However, the multistage principle is mostly a stopgap solution, and of little economic value if the efflux velocities are insufficient.

Indeed, if one applies this multistage principle of propulsion by superimposing several aircraft one on top of the other, the total cost increases sharply and the procedure can in many instances become impossible owing to its lack of economy. In cases where economy plays no role—for instance in the event of a first flight through space—it can prove to be an effective means of attaining this goal with moderate efflux velocities.

One way of improving the economic aspect is to place the first stage, in the form of a supersonic rocket catapult, on the ground.

If the aircraft is considered as second stage, with $G_1/G_2 = 10$; $c_2 = 3000$/-sec; $k_2 = 0.8$; and the first stage as a catapult with G_0 $(G_1 + \Delta G_1) = 1.43$; $c_1 = 1500$ m/sec; and $k_1 = 0.92$; the end speed after the first stage will be $k_1c_1 \ln G_0/(G_1 + \Delta G_1) = 500$ m/sec and after the second stage $v_{1_{st}} = 500 + k_2c_2 \ln G_1/G_2 = 6000$ m/sec against only about $v_1 = 5500$ m/sec without catapult. According to section on rocket engines of this article, the ranges in both cases are in the ratio of 17,480 km to 13,400 km; the gain due to the catapult therefore amounts to about 30 per cent and can still be more for higher values of c_2.

The entire flight then proceeds roughly as follows: The rocket aircraft is catapulted along a horizontal ground track of 3 km by means of powerful rockets functioning during approximately 11 sec, until the aircraft has gained about $1^1/_2$ times sonic speed; subsequently, with a trajectory forming an angle of 30 degrees with the horizon (this inclination will later diminish) the aircraft rises under full thrust of its rockets, to attain altitudes of 50,000 to 150,000 meters and end speeds which are multiples of the efflux velocity. This ascent lasts 4 to 8 min, during which time the fuel is consumed. At the end of the ascent, the rocket stops and the aircraft continues

its course along a sort of wave-shaped trajectory with oscillations of decreasing amplitude, solely on its kinetic and potential energy.

Because of its wings and the supporting properties of its fuselage, the aircraft, descending along a ballistic curve, will bounce on the lower layers of the air and thus be thrown upward again, just as a flat stone ricochets when thrown along the surface of a pond; each plunge into the denser air will result in a part of the kinetic energy being consumed, so that the initially long jumps will gradually become shorter, finally to transform into an even gliding flight. At the same time, along the many thousands of kilometers of gliding trajectory, the flying speed will decrease from its high initial value to the normal landing speed.

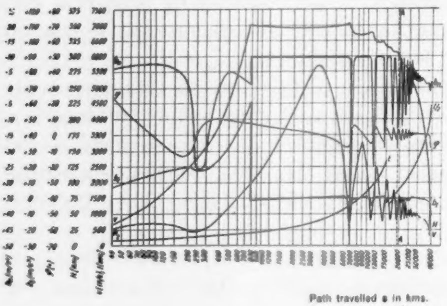

FIG. 21 FLYING SPEED v, FLIGHT ALTITUDE H, TRAJECTORY INCLINATION ϕ, TANGENTIAL ACCELERATION b_t, NORMAL ACCELERATION b_n, AND ELAPSED TIME t OF THE TRAJECTORY OF A ROCKET BOMBER WITH c—4000 M/SEC, v_0—7000 M/SEC AND BOMB-LOAD B—3.8 TONS DROPPED FROM 40,000-METER ALTITUDE (A—DROPPING POINT) AT 6060 M/SEC FLYING SPEED AND 19,200 KM DISTANCE FROM THE TAKE-OFF POINT.

Enormous flight trajectories of this kind, passing along the surface of the earth, are shown in Fig. 21 in exaggerated scale. As regards the length of terrestrial long-range flight trajectories, these can be estimated by means of the relation contained in Section C 1.

Domains of Application of Atomic Ramjet or Atomic Rocket Aircraft: The considerations made so far have shown that both chemically fueled ramjet aircraft and chemically fueled rocket aircraft enable ranges of the order of 10,000 km to be attained which may be even much more with chemical rocket aircraft.

At the same time, the operating costs and the flying speeds would seem to attain higher levels in the case of rocket aircraft.

With the introduction of atomic energy, the ranges of both these aircraft types will become unlimited, though this will be for entirely different reasons.

In the case of the ramjet aircraft, the fuel consumption will be practically nil; at speeds which one will be able to vary arbitrarily between $v/a = 0.6$ and 3.5, and at altitudes below 30,000 meters, such an aircraft will fly as long as one wishes, being entirely maneuverable due to the utilization of the atmospheric air as a means of propulsion; its flight will be characterized by the high proportion of transportable pay load and the modest requirements as regards technical and organizational installations. It will represent a means of transportation which is in a sense earthbound; as an in-

strument of warfare, it would be under the constant menace of hostile fighter aircraft and antiaircraft defenses.

For the pure rocket aircraft, on the other hand, the introduction of atomic energy in no way signifies the elimination of the masses to be taken on board; if a trajectory becomes unlimited, this is alone due to the extremely high flying speeds, which enable the aircraft to emerge definitely from the earth's atmosphere. Once it has started, the flight will only be able to be controlled within very narrow limits, since it will proceed according to the laws of motion of a celestial body. The pay load will invariably represent a small part of the gross weight; on the other hand, operations will demand considerable technical and organizational installations. It would seem to represent primarily an outer-terrestrial means of communication, constituting a step along the road to interstellar flight; as an instrument of warfare, it will probably be outside the scope of enemy action.

Economic Aspects

In the assessment of the economic aspects of an aircraft, the usual computation of the operating costs is of essential importance only in connection with civil aviation, but is of no significance at all where military or scientific standpoints are concerned.

It is above all in the domain of scientific research that the ratio between the initial costs and the result to be obtained become intricate, since even the most farsighted and cognizant of scientists is unable to predict the probability of the goal being reached.

The development of space weapons, the so-called V-weapons, during the war in Germany, serves as an example of the considerable difficulty offered in this connection.

From the general point of view, the V-1 represented a particularly rational solution of the problem of transporting a given quantity of explosives over a given distance, coupled with a minimum of work and human risk. The degree of efficiency reached would have been still greater, however, had these devices attained the speeds and altitudes anticipated.

One cannot straightway ascribe a similar degree of efficiency to the V-2 in the stage of development it had reached when put into wartime operation, since the manufacture and operation of these rockets required far more work and material per ton-kilometer of load carried, and these factors were not offset by these rockets' immunity to attack by hostile forces.

Considered from another angle, the development of these two weapons, which consumed a considerable portion of the nation's intellectual and manual potential, appears as having been a waste in so far as by the time they were ready for operation the need was not so much for offensive weapons as for a means of defense against air attacks, and the very potential which would have enabled such defensive weapons to be developed, had been expended on the V-1 and V-2.

Looking at it from another aspect, the development of the liquid-fuel rockets of Oberth appears to have been more advantageous. If these rockets had really constituted the beginning of the development of interstellar navigation, then the enormous expenses of all kinds involved in the development of the V-2 appear in an entirely different light.

The Problem of the Fighter Aircraft

The problem of the fighter aircraft furnishes a better example of what economy in military aviation can signify. Basing our views on the technical situation existing in Germany during the last years of the war, we see that the goal of war economy also tends in general to safeguard the national potential in man-hours, and to use it in a rational manner.

The mission of the fighter aircraft is to destroy enemy aircraft. If the number of aircraft destroyed is proportional to the hours flown, the total of the man-hours per efficacious hour flown constitutes a given value which could serve as a basis for comparing different types of fighter aircraft.

If one furthermore disregards the costs which are common to every type of fighter, in other words the high costs incurred by the crew, ground organization, and armament, we then come to the specific supplementary costs coupled with each type (airframe, engine, fuel) presented in Fig. 22, expressed in man-hours per efficacious hour flown in combat against the enemy, in function of the life duration of fighter aircraft.

Since each kilogram of aircraft perhaps represents a hundred times the number of man-hours pertaining to a kilogram of fuel, expensive aircraft with low fuel consumption (turbojet fighters, propeller fighters) can only be more economic than cheap aircraft with high fuel consumption (rocket or ramjet fighters) if they have a longer life duration.

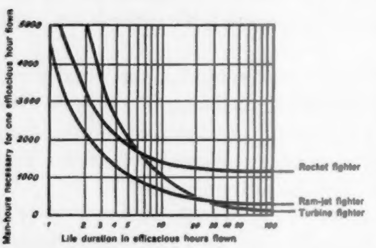

FIG. 22 ECONOMIC COMPARISON OF FIGHTER AIRCRAFT, DEPENDING ON THEIR LIFE DURATION IN EFFICACIOUS HOURS FLOWN.

In view of the fact that the life duration is especially short in the case of fighter aircraft, the price of the fuel consumed during this short life, even if the specific consumption is high, plays but a small role when compared with the price of manufacturing the aircraft. The low manufacturing price of a ramjet fighter, for example, can therefore be fully exploited in this connection. The still topical problem of unmanned aircraft in operation against terrestrial and aerial targets also comes within the scope of this examination of the economic aspects of combat aircraft.

From a purely economic point of view, the losses in human life have much heavier consequences in wartime than the heaviest material losses.

These facts are the principal cause of the progressive mechanization of war, which tends to diminish the number of individuals exposed to risk; and these facts justify the utilization of unmanned aircraft.

Summary

The prospects of jet-reaction flight are manifold and interlaced. Four principal ways ahead are discernible, and these are presented by four types of prime mover: resonance ducts, turbojets, ramjets, and rockets.

Looking ahead, the prospects of the resonance duct seem to become vague and to lose themselves along numerous small paths.

After a number of side roads which will continue to be trodden, the prospects of turbojets, it would seem, must ultimately merge with those of the ramjet; and this latter engine type will probably provide the ideal means of propulsion for high-speed airliners encircling the globe.

The fourth way ahead, that of the rockets, seems to be destined to open to humanity the doorway to outer-terrestrial navigation and interstellar flight.

POTENTIALITIES AND DEVELOPMENT PROBLEMS OF THE TURBOPROP ENGINE

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Introduction

THE gas-turbine power plant employing a jet for producing a thrust, called the turbojet engine, has already demonstrated its suitability for propelling aircraft at speeds above 500 mph. At flight speeds around 400 mph, the propulsive efficiency of jet propulsion is of the order of 40 per cent while a propeller may be expected to have a propulsive efficiency of the order of 80 per cent. Analyses have demonstrated that, due to the much higher propulsive efficiency of the propeller, the gas turbine driving a propeller, called the turboprop engine, is considerably more efficient than the turbojet engine for continuous flights in the 400–500 mph speed range. For long-range aircraft, the savings in fuel to be expected from the higher efficiency of the turboprop engine will more than compensate for the greater weight of the turboprop engine plus propeller.

Fig. 1 is an estimate, taken from (10)[1], of the probable fields of application of the piston engine and the turboprop engine. The figure shows that for high-altitude operation and speeds in the 400–550 mph range, the turboprop engine will dominate the large-aircraft propulsion field after it has been fully developed and adequately tested in service.

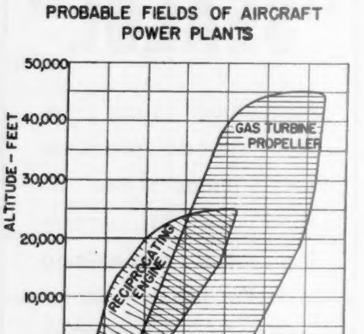

FIG. 1 PROBABLE FIELDS OF APPLICATION OF AIRCRAFT POWER PLANTS

Fig. 2, taken from (10), compares estimated power losses at different flight speeds for the same airplane propelled with piston engines and turboprop engines. It is apparent that the large reduction in nacelle losses greatly improves the net-power output at high flight speeds with turboprop-engine propulsion.

Presented at a meeting of the New York Section of the American Rocket Society, New York, N. Y., Jan. 21, 1949.

[1] Numbers in parentheses refer to Bibliography on page 128.

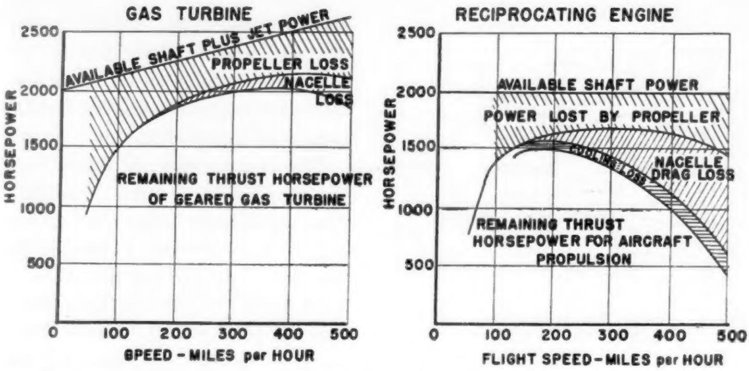

FIG. 2 ESTIMATE OF TYPICAL FLIGHT LOSSES FOR PISTON ENGINE AND TURBOPROP POWER PLANTS

Fig. 3 presents the expected performance characteristics, based on factory test, of the General Electric Type TG-100 turboprop engine (12). The take-off rating of this engine is approximately 2400 hp and its rating at 500 mph and 25,000 ft altitude is 2050 equivalent shaft horsepower (shp). The weight of the complete power plant including reduction gear and all accessories is 1975 lb, and its fuel consumption at 500 mph and 25,000 ft altitude is expected to be 0.51 lb per shp-hr.

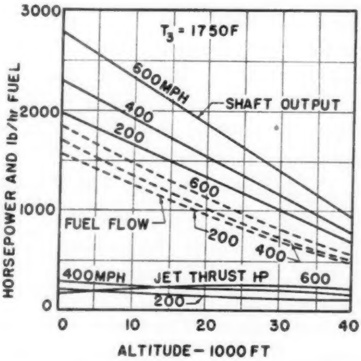

FIG. 3 PERFORMANCE CHARACTERISTICS OF TG-100 AT NORMAL RATING

To obtain an appreciation of the potentialities and design flexibility of the turboprop engine, it will be advantageous to review some of the performance characteristics of the simple open-cycle gas-turbine power plant, hereafter termed the basic plant.

The Basic Plant

Fig. 4 illustrates schematically the essential elements of the basic plant and its counterpart, a turboprop engine equipped with an axial-flow compressor. The major components are: air compressor, combustion chamber or combustor, turbine, and control system. The air compressor raises the pressure of the air entering the power plant to several times its entrance value. The magnitude of the pressure ratio is governed by the application. The air compressor may be either of the axial flow, centrifugal, or positive

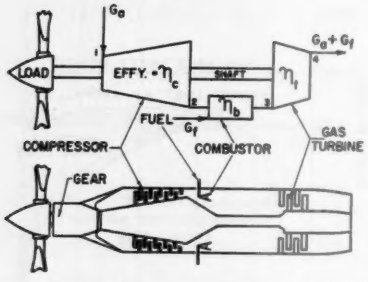

FIG. 4 ESSENTIAL ELEMENTS OF AN AXIAL FLOW TURBOPROP ENGINE

displacement type; however, the aircraft applications favor the axial type because of its larger air-induction capacity, smaller frontal area, and higher efficiency.

The compressed air flows from the compressor to an internally fired combustion chamber wherein fuel is burned with a portion of the air. The balance of the air is subsequently mixed with the combustion products to lower the temperature of the gases entering the turbine to a safe value for the turbine blading and to give a uniform temperature distribution. Current practice achieves turbine-inlet temperatures ranging from 1200 F to 1800 F depending upon the desired life for the power plant. In stationary and marine plants, where long life is the primary requirement, the maximum temperature is limited to approximately 1200 F. In aircraft applications where life can be sacrificed to the interests of low power-plant weight and fuel economy, the temperature of the gases may be as high as 1750 F for continuous operation and 1900 F for 15-minute periods (12).

The hot gases, mainly highly heated air admixed with a small amount of combustion products, expand in the turbine and develop power. The greater portion of the turbine output is consumed in driving the air compressor, and the balance constitutes the net power available for driving the external load, such as propeller, and auxiliaries; also for overcoming the parasite losses. The pressure drop in the turbine element of the turboprop engine is larger than that for a turbojet engine. Consequently, the gases leaving the exhaust nozzle of the turboprop engine can furnish only a small jet-propulsion thrust. The proportioning of the shaft power and jet thrust is at the disposal of the engine designer.

General Characteristics of the Basic Plant

Since the complete characteristics of the basic plant have been adequately described in the literature, only the characteristics pertinent to the present discussion will be reviewed.

The Ideal Turboprop Engine: If it is assumed that there are no parasite losses, that the compressor and turbine are 100 per cent efficient, that the working fluid is air, and that the inlet diffuser pressure ratio equals that of the exhaust nozzle, then the power plant operates on the Brayton cycle, and its efficiency is a function only of the cycle pressure ratio.

Fig. 5 presents the air-cycle efficiency of the ideal turboprop engine as a function of the cycle pressure ratio. It is seen that the air-cycle efficiency is

independent of the turbine-inlet temperature. Accordingly, the efficiency of the ideal engine can always be raised by increasing the cycle pressure ratio.

Fig. 6 presents the specific work B/lb of the ideal engine, as a function of the cycle pressure ratio. It is seen that for a fixed turbine-inlet temperature, the specific work increases rapidly at first with the pressure ratio, reaches a maximum value, and then decreases. Furthermore, as the turbine-inlet temperature is raised, the pressure ratio for the maximum specific work increases.

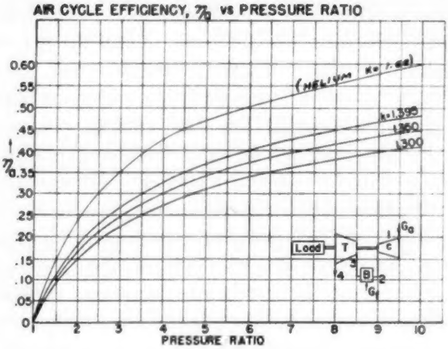

FIG. 5 AIR CYCLE EFFICIENCY VERSUS PRESSURE RATIO (BASIC CYCLE)

Figs. 5 and 6 show that for the ideal engine the thermal efficiency or specific fuel consumption is independent of the turbine-inlet temperature, but that the output per lb of the working fluid depends directly upon the temperature rise between the compressor and turbine inlet. The higher the permissible turbine-inlet temperature, the larger must be the pressure ratio if the maximum possible work is to be obtained from the engine. The aforementioned characteristics pertinent to the ideal power plant are only indicative of the relative importance of turbine-inlet temperature and pressure ratio on the performance of an actual turboprop engine.

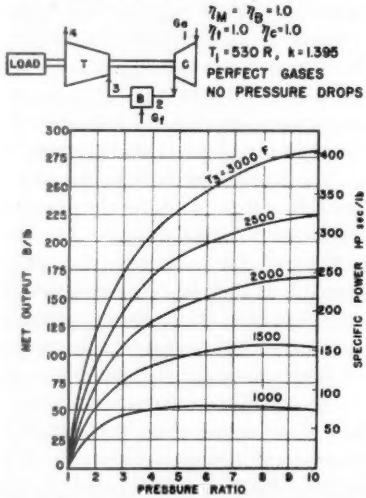

FIG. 6 NET OUTPUT VERSUS PRESSURE RATIO FOR DIFFERENT TURBINE-INLET TEMPERATURES

Effect of Machine Efficiency: The ratio of the specific power obtained from the engine to the total power developed by the turbine element is known as the work ratio. All gas-turbine power plants are characterized by the fact that their work ratios have low values: of the order of 0.2 to 0.6 depending upon the design of the plant. The low work ratio is due to the large power consumption of the air compressor, and the higher values are obtained by complicating the plant design with

the addition of heat exchangers, reheaters, and intercoolers. But these modifications to the basic plant do not appear to be readily adaptable to the turboprop engine. For current turboprop engines, the work ratio is close to 1/3, which means that two of every horsepower developed by the turbine element are consumed in driving the air compressor, and makes available a net work of one horsepower for driving the propeller. Consequently, a 5 per cent reduction in the efficiency of the turbine element results in a 15 per cent reduction in the net output, and a 5 per cent reduction in compressor efficiency decreases the net output by 10 per cent. It is apparent, therefore, that the turbine efficiency, denoted by η_t, and the compressor efficiency, denoted by η_c, greatly influence the net output and thermal efficiency of the real engine. Furthermore, the lower the efficiencies of the turbine and air compressor, the higher must be the turbine-inlet temperature to produce a given net output. The interdependence between turbine-inlet temperature, net output, and the machine efficiency (defined as the product of the compressor efficiency η_c and the turbine efficiency η_t), is a fundamental characteristic of the gas-turbine power plant.

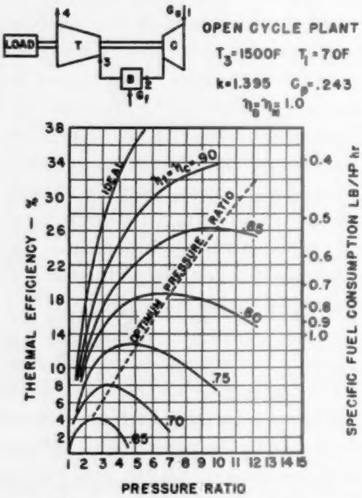

FIG. 7 EFFECT OF MACHINE EFFICIENCY ON THERMAL EFFICIENCY FOR A FIXED TURBINE-INLET TEMPERATURE

Fig. 7 illustrates, for a turbine-inlet temperature of 1500 F, how the thermal efficiency varies with the cycle pressure ratio for different values of the machine efficiency $\eta_t \eta_c$. It is seen that increasing the machine efficiency markedly decreases the specific fuel consumption if the pressure ratio is simultaneously increased. But there is an optimum ratio for each value of machine efficiency and turbine-inlet temperature.

Fig. 8 presents the specific output as a function of the cycle pressure ratio for different values of machine efficiency, for a turbine-inlet temperature of 1500 R. It is seen that in this case too, there is an optimum pressure ratio. But the optimum pressure ratio for the maximum net work per lb corresponding to a given value of machine efficiency is always lower than the optimum pressure ratio corresponding to the maximum thermal efficiency. It is further noted that for values of machine efficiency exceeding 0.75 approximately, the net work per lb of working fluid is influenced to a smaller degree by raising the pressure ratio than is the thermal efficiency.

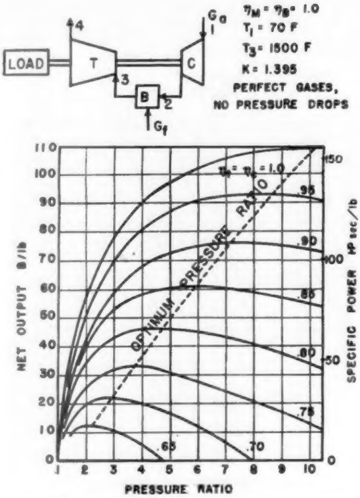

FIG. 8 EFFECT OF MACHINE EFFICIENCY ON NET OUTPUT FOR A FIXED TURBINE-INLET TEMPERATURE

The values presented in Figs. 7 and 8 are somewhat higher than those to be expected from an actual engine because the effects of parasite losses and variation of the specific heat of the working fluid with temperature were not considered.

Effect of Pressure Loss: Pressure losses adversely affect both the specific output and thermal efficiency of a gas-turbine power plant. Care must, therefore, be exercised to keep the pressure losses at a low value. In general, pressure drops occur in the piping leading from the compressor to the combustion chamber, in the combustion chamber, and in the exhaust passages from the turbine. The deleterious effect of large pressure losses on both the specific output and thermal efficiency is illustrated in Fig. 9.

The problem of designing for low pressure loss is aggravated in applications such as aircraft where small frontal area and low weight are of utmost importance. Because of the small cross-sectional area of the flow passages, high gas velocities, with their attendant large pressure losses, are difficult to avoid. The final design in any type of gas-turbine power plant is inevitably a compromise between the space and weight limitations and the allowable pressure losses.

Effects of Turbine Inlet Temperature: Fig. 10 presents the thermal efficiency as a function of the cycle pressure ratio for different values of turbine-inlet temperature and the conditions presented in the figure. It is seen that to obtain the maximum possible thermal efficiency corresponding to any assigned value of turbine-inlet temperature, the cycle pressure ratio must be adjusted accordingly.

Fig. 11 is a chart from which the optimum pressure ratio can be deter-

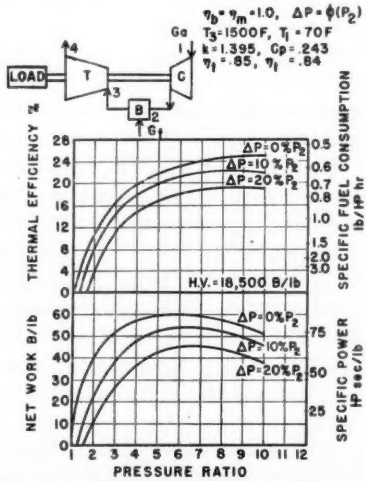

FIG. 9 EFFECT OF PRESSURE DROP ON THERMAL EFFICIENCY AND NET WORK

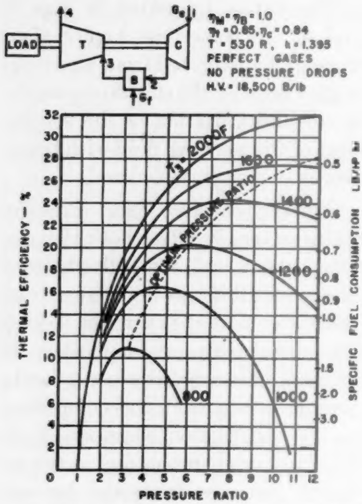

FIG. 10 THERMAL EFFICIENCY VERSUS PRESSURE RATIO FOR DIFFERENT TURBINE-INLET TEMPERATURES

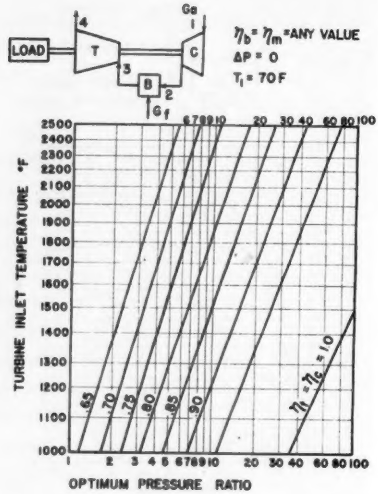

FIG. 11 TURBINE-INLET TEMPERATURES VERSUS OPTIMUM PRESSURE FOR DIFFERENT VALUES OF MACHINE EFFICIENCY

mined for different values of turbine-inlet temperature and machine efficiency, assuming that the compressor and turbine are equally efficient. It is seen that if the turbine-inlet temperature is increased and the machine efficiency is held constant, the pressure ratio must also be increased if the maximum work is to be obtained from the engine.

If the machine efficiency is raised while the turbine-inlet temperature is held constant, this also calls for increasing the pressure ratio. Both Figs. 10 and 11 neglect the losses due to pressure drops and the variation of specific heat with temperature. They illustrate quite closely, however, the interrelation between the different variables.

Fig. 12, taken from (14), presents similar information but also takes into consideration the aforementioned neglected factors, and in addition shows the effect of operation at altitude. The improvement in specific fuel consumption at altitude is due to the reduction in compressor work resulting from the lower intake air temperature. It is seen that the pressure ratio for minimum fuel consumption is 20 to 1 for a turbine-inlet temperature of 2000 R and must be raised to 60 to 1 for 3000 R. With a turbine-inlet temperature of 3000 R, the fuel consumption is reduced only from about 0.36 to 0.33 lb per bhp-hr by increasing the pressure ratio from 20 to 1 to 60 to 1.

Fig. 13 presents the net work as a function of pressure ratio for different turbine-inlet temperatures for the same conditions as in Fig. 10. It is seen that as the inlet temperature is raised, the net work becomes less sensitive to changes in pressure ratio, but there is an optimum pressure ratio for each

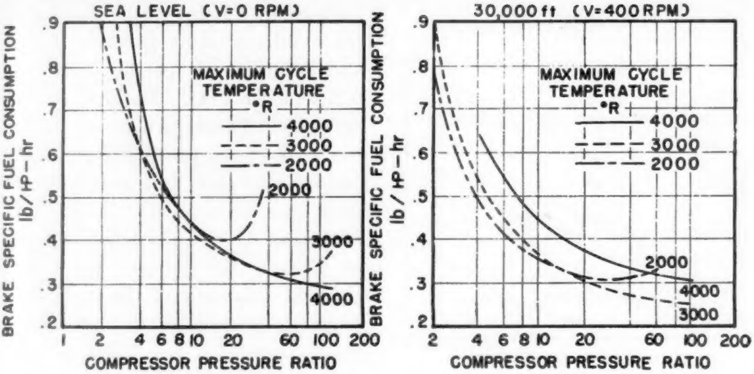

FIG. 12 EFFECT OF ALTITUDE AND TURBINE-INLET TEMPERATURE ON THE SPECIFIC FUEL CONSUMPTION

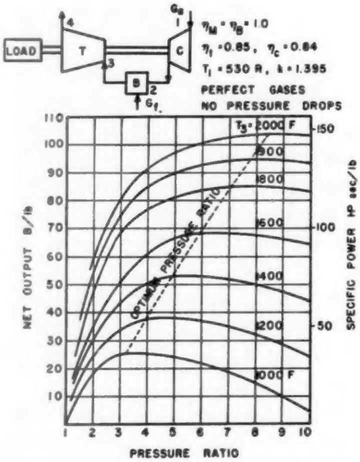

FIG. 13 NET OUTPUT VERSUS PRESSURE RATIO FOR DIFFERENT TURBINE-INLET TEMPERATURES

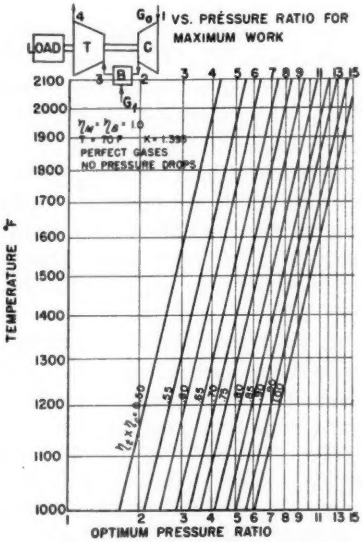

FIG. 14 TURBINE-INLET TEMPERATURE VERSUS OPTIMUM PRESSURE RATIO FOR DIFFERENT VALUES OF MACHINE EFFICIENCY

inlet temperature. Fig. 14 presents the turbine-inlet temperature as a function of the optimum pressure ratio for different values of machine efficiency. The curves presented in Figs. 13 and 14 neglect all parasitic losses and the effect of the variation of specific heat with temperature. The values presented in the figures are only approximately correct.

Fig. 15 illustrates the effect of altitude on the specific power for different

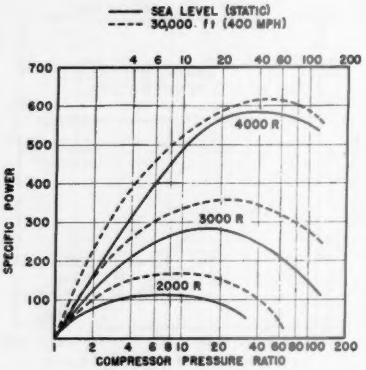

FIG. 15 EFFECT OF ALTITUDE AND TURBINE-INLET TEMPERATURE ON THE SPECIFIC POWER

turbine-inlet temperatures and also takes into account parasitic losses and variation of specific heat with temperature (14). Because of the lower air-inlet temperature at outlet, the work of compression is reduced, and for a constant turbine-inlet temperature, the net work at altitude is increased. The beneficial effects of altitude on the net work and thermal efficiency of the turboprop engine indicate that the best performance results if the flight is conducted at high altitude.

The choice of turbine-inlet temperature is related to the desired operating life for the power plant, and with conventional designs is limited by the available metals. For any metal there is a relationship between the duration of the applied stress, allowable deformation, and its operating temperature. The influence of the aforementioned time factor must be given prime consideration in designing the plant. The allowable stresses for the parts to be exposed to high temperatures for long periods of time must be limited to a value which will preclude the plastic deformation or flow of the highly heated metal exceeding a certain maximum value during the desired operating life for the power plant. In general, the allowable dimensional increases with service time are limited to between one-half of one per cent and two per cent.

Consequently, to take advantage of the large power output and reasonable thermal efficiencies which are potentially available from the use of higher inlet temperature, methods for cooling the heated parts, particularly the turbine blades and disk, must be developed.

Potentialities: From the discussions already presented, it is apparent that the future of the turboprop engine is closely related to the engineering development of more efficient compressors, turbines, and means for making it possible to operate with higher turbine-inlet temperatures. The influence of machine efficiency on performance has been indicated, and it can be said generally that for each one per cent of improvement in turbine efficiency, the cycle efficiency is raised approximately three per cent. Efficient compressors and turbines are currently available, but since it is believed that they have not as yet attained their ultimate peak efficiencies, it is reasonable to believe that the future will bring more efficient air compressors and turbines.

The importance of employing higher turbine-inlet temperatures cannot be overemphasized because of the relationship between the turbine-inlet

temperature, the net output, thermal efficiency, and machine efficiency. A factor which should not be viewed too lightly is that both the compressor, turbine, and metallurgical developments can proceed along fairly well-established engineering lines since they are independent of fuel technology. This is not the case for spark-ignition or compression-ignition internal-combustion engines.

Fig. 16 compares the gains which can be effected by developments to improve compressor efficiency compared with those obtainable with higher turbine-inlet temperatures (13). Fig. 17 compares the gains obtainable with further improvement in turbine efficiency with those attendant to raising the turbine-inlet temperature. It is apparent that the successful prosecution of developments which enable the turbine-inlet temperature to be raised will pay handsome dividends.

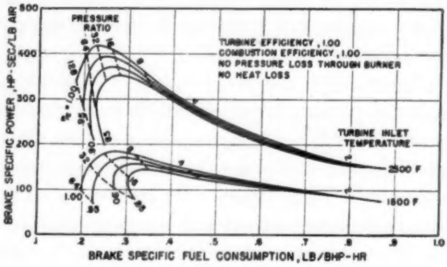

FIG. 16 EFFECT OF COMPRESSOR EFFICIENCY AND TURBINE-INLET TEMPERATURE ON PERFORMANCE

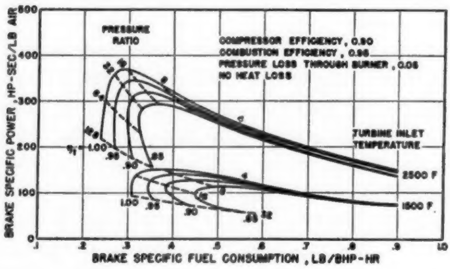

FIG. 17 EFFECT OF TURBINE EFFICIENCY AND TURBINE-INLET TEMPERATURE ON PERFORMANCE

Metallurgy: The great benefits accruing from the ability to employ higher temperatures at the turbine inlet have been discussed. The accomplishment of that objective has until recently been dependent entirely upon the successes of research in the field of high-temperature metallurgy. Several alloys capable of withstanding dynamic loads under highly stressed conditions and with satisfactory creep characteristics at the elevated temperatures have been developed in the past ten years, and several research organizations are actively engaged in developing better alloys. In view of the accomplishments made during the past ten years, it is not too optimistic to expect that the next ten years will bring forth further improvements.

Metallurgical advancements by themselves, however, do not appear to have the potentiality of making it possible to utilize turbine-inlet temperatures of the order of 2000 F to 3000 F. Operation at such temperature levels must be based on successful methods for cooling the turbine blades and disk. The

National Advisory Committee for Aeronautics at the Lewis Flight Propulsion Laboratory is conducting basic research concerned with the problems of blade cooling (13). The results obtained so far encourage one to believe that operation with turbine-inlet temperatures which are much higher than normal will become a possibility in a few years.

Air Compressors: The most widely used type of air compressor for general purposes is the centrifugal compressor, and its application to the turbo-jet engine has resulted in the development of high-speed lightweight units.

Unfortunately, the pressure ratios which can be developed by the centrifugal compressor impeller appear to be limited to approximately 4 to 1 if reasonable efficiencies are to be obtained. To obtain higher pressure ratios, multistaging is necessary, which increases the weight and the size of the air-compressor installation. Furthermore, to reduce the power consumption of a multistage centrifugal compressor, intercooling between the stages is essential, which further increases the weight and complexity. The flow processes in the impeller of the centrifugal compressor are not fully understood, and research is needed to clarify them. From the results of such research, methods for achieving higher pressure ratios with good efficiencies may result.

Currently the axial flow compressor is the most attractive for turboprop engines. Efficiencies of 85 per cent have been reported for production units, and 90 per cent for experimental units. But this machine has the disadvantage that its pressure ratio per stage is low, of the order of 1.05 to 1.15 depending upon the blading design. To produce reasonable pressure ratios several stages are required, which tends to make the axial flow compressor long and heavy. This is particularly true when unsymmetrical blading is used. The axial flow compressor is receiving intensive development to raise the pressure ratio per stage, reduce its weight and size, and improve its efficiency.

To achieve the high pressure ratios which are required with higher turbine-inlet temperature, the possibilities of a compressor plant consisting of an axial flow compressor followed by a centrifugal compressor needs fuller exploration.

Turbine: The aerodynamic and thermodynamic problems connected with the design of an efficient turbine that is to operate on a heated gas, do not appear to be serious or unusual, apart from the metallurgical problems inherent to the utilization of high temperatures. To obtain the efficiencies and high specific outputs resulting from the higher turbine-inlet temperatures will require research in cooling methods as already explained. For uncooled turbine blades operating at temperatures of the order of 1200 F, materials having sufficiently high endurance strength, low enough creep rate, and sufficient oxidation resistance have been developed, and a wealth of service experience has been accumulated at that temperature. The

service experience with the alloys for turbine blades which are to operate at temperatures up to 1800 F is not as extensive. As metallurgical advances are made, the possibility of employing uncooled blades at temperatures somewhat higher than 1800 F for short-life engines, such as the turboprop engine, may become a reality. In this same connection, the successful application of ceramic materials to the highly heated and stressed metal parts of the engine can contribute to making higher operating temperatures feasible. At the moment, however, the achievement of that objective by cooling the blades and disks with high pressure air or a liquid coolant appears more promising.

There should be no unsurmountable problems in developing a turboprop engine of low specific weight to operate with the higher pressure ratios required to obtain the maximum performance inherent to the use of higher turbine-inlet temperatures.

Fig. 18 presents an estimate made by the NACA of the specific weight of the turboprop engine as affected by the turbine inlet temperature (14). The figure also presents estimated values of specific fuel consumption and range. In making the calculations, it was assumed that the gear-box and propeller weight amounts to 0.6 lb per hp and that the propeller efficiency was 80 per cent. In addition, weight allowances were made to account for liquid cooling of the turbine parts.

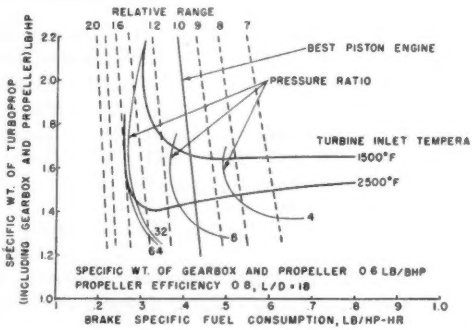

FIG. 18 ESTIMATE OF THE EFFECT OF TURBINE-INLET TEMPERATURE AND PRESSURE RATIO ON THE SPECIFIC WEIGHT OF TURBOPROP ENGINE; ITS SPECIFIC FUEL CONSUMPTION AND RELATIVE RANGE

There are, of course, problems of thermal distortion, but they are, in large part, design problems requiring careful analysis of the contributing factors for their solution. Thus, rapid changes in the thickness of metal sections, especially at connections to piping, should be avoided. The expansion and contraction phenomena must be given careful study, and the design so arranged that it precludes the possiblity of the parts being damaged by these movements. Where possible, the parts subjected to thermal stresses should be segregated from those which are stressed mechanically.

Conclusions

The turboprop engine has been developed to the stage where it is being subjected to testing and is being considered for the propulsion of aircraft at

speeds in the 400–500 mph range. Its flexibility is such that it can be designed to meet the requirements of a variety of services. It holds forth the promise of combining high thermal efficiency, low weight, and small space requirements. Its further improvement will depend upon the success of research and development directed toward making it possible to employ higher turbine-inlet temperatures, with higher cycle pressure ratios, with moderate improvements in machine efficiency. Since no turboprop engines are being built on a production basis at this time, statements regarding their cost per installed horsepower cannot be significant.

Acknowledgment

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References

1 "The Gas Turbine, Its History, Development and Prospects," by Adolph Meyer, Proceedings, The Institution of Mechanical Engineers, 1939, vol. 141, p. 197.

2 "The Basic Gas Turbine Plant and Some of Its Variants," by J. K. Salisbury, *Mechanical Engineering*, June, 1944, vol. 66, p. 375.

3 "The Gas Turbine; I, II, III," by J. T. Rettaliata, *Allis-Chalmers Electrical Review*, Sept., 1941, p. 20; Dec., 1941, p. 27; March, 1942, p. 26.

4 "Gas Turbine Developments Portend Power Changes," by E. A. Fisher and C. A. Meyer, *Industry and Power*, May, 1944, p. 64.

5 "Value of Wet Compression in Gas Turbine Cycles," by R. V. Kleinschmidt, *Power*, March, 1947.

6 "The Gas Turbine as a Possible Marine Prime Mover," by C. R. Soderberg and R. B. Smith, *The Society of Naval Architects and Marine Engineers*, Nov. 11, 1943, p. 115.

7 "The Escher Wyss-AK Closed-Cycle Turbine, Its Actual Development and Future Prospects," by C. Keller; Transactions of The American Society of Mechanical Engineers, Nov., 1946, p. 791.

8 "The Gas Turbine in Industry," by J. R. Carlson, presented before SAE, March 14, 1946, Washington, D. C.

9 "A Compact Locomotive-Type Gas Turbine," by T. J. Putz; *Westinghouse Engineer*, March, 1947, pp. 35–39.

10 "The Place of the Gas Turbine in Aviation," by F. W. Godsey, Jr., and C. D Flagle, *Westinghouse Engineer*, July, 1945, pp. 121–127.

11 "A Marine Gas Turbine Plant," by C. R. Soderberg, R. B. Smith and Lt. Comdr. A. T. Scott, The Society of Naval Architects and Marine Engineers, No. 6, Dec., 1945, p. 249.

12 "An Aircraft Gas Turbine for Propeller Drive," by A. Howard and C. J. Walker, *Mechanical Engineering*, Oct., 1947, p. 827.

13 "The Advantages of High Inlet Temperature for Gas Turbines and Effectiveness of Various Methods of Cooling the Blades," by O. W. Schey, Paper No. 48-A-105. Presented ASME 1948 Annual Meeting, Nov. 29, 1948.

14 "Performance Possibilities of the Turbo-Propeller," by R. O. Bullock and R. E. English, Preprint No. 147, IAES, 1948.

SUPERSONIC GUIDED MISSILES

By R. E. Gibson

DURING the past decade rockets and guided missiles have emerged from the status of figments of fantasy or, at best, scientific curiosities to occupy an important position in serious technical thought and effort. Today, many tens of millions of dollars and the services of thousands of scientists and engineers are being devoted to the development of jet-propelled automatically-controlled high-speed missiles. The subject is so large, complex, and fluid that a long course of lectures would be required to give a comprehensive account of the developments that are taking place. Any attempt to cover in one lecture even those parts of the subject that are not under security classification, would leave the audience in a state of frustrated bewilderment. This paper will be limited to a brief survey of the general technical problems posed by the guided missile, a short account of an organization to solve these problems, and finally a more detailed account of some of the propulsive systems being developed for guided missiles with special emphasis on rockets and ramjets.

Jet-propelled missiles are not new in the armories of nations. Authentic records[1] show that the Chinese used "arrows of flying fire" against the Mongols in the seige of Kai-Fung-Fu in A.D. 1232. Rockets were also known in Europe by that time and found considerable military use in the Middle Ages. In 1780, or thereabouts, a revival in the use of rockets took place in India and the British suffered badly from rocket attacks at the battle of Seringapatam in 1792. Shortly thereafter, 1800–1805, Col. William Congreve, M.A., introduced a radical improvement into the manufacture of rocket propellants thereby increasing the range to 2500 yd. This led to the extensive use of artillery rockets, and by 1830 there was hardly a nation whose military establishment did not contain rocket-artillery units. About this time, 1846, an American named William Hale introduced a further improvement by placing small inclined vanes in the rocket nozzle which caused it to spin rapidly and thereby gain gyroscopic stability. However, the rapid improvement in guns during the middle of the last century caused a decline in the popularity of artillery rockets, and the last rocket unit, an Austrian one, was disbanded in 1866.

Rockets were not used in World War I, but toward the end R. H. Goddard and his assistants in the United States introduced double-base smokeless powder as a propellant and produced a more powerful rocket than had hitherto been available. The development was dropped completely at the

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Presented at a joint meeting of the Institute of Aeronautical Sciences and the AMERICAN ROCKET SOCIETY, New York, N. Y., Feb. 8, 1949.

[1] "Rockets and Space Travel," by Willy Ley, Viking Press, Inc., New York, N. Y., 1947, chap. 3.

close of the war and not revived officially until 1940. With the aid of double-base smokeless-powder propellants the British and the Germans developed powerful artillery rockets between 1930 and 1940. These solid-fuel rockets together with American and Japanese versions were used extensively in World War II from installations mounted on ground, ships, and airplanes, and piloted the new era of jet-propelled missiles.

In the second decade of this century the development of rockets propelled by liquid fuels began in a small way through the enthusiasm of isolated amateur groups in various countries. An American group under Dr. Goddard first developed, in 1926, a usable rocket powered by liquid oxygen and gasoline, and a German group proceeding in ignorance of this work was successful with the same fuel combination in 1930.

About 1936 the German rocket work was well established under government auspices, and a development was started that led to the gigantic V-2 rocket with its range of 200 miles and its extremely effective accuracy. The seed, sowed by the enthusiasm of the early rocket societies and forced to rapid growth in the hothouse of wartime necessity, had borne magnificent if terrifying fruit.

World War II also brought to maturity a number of other technical advances which had been incubating during the previous decade, notably turbojets for the high-speed propulsion of aircraft, radar with its techniques for generating high-energy pulses, measurement of short-time intervals and automatic control for measuring precisely range and direction, forms of complex communications by electromagnetic waves, loran navigation systems, small and rugged electronic equipment culminating in radio proximity fuses, high-speed electronic and mechanical computers and servomechanisms for interpreting signals and producing rapid automatic response, high-speed and long-range airplanes, and finally atomic bombs.

Through these developments man not only gained control over a source of energy millions of times more powerful than ever before available, but also developed techniques to supplement human intelligence, skill, and effort by automatic devices and thereby increased by orders of magnitude the range, speed, and precision of his destructive activity. It is no wonder that many people predicted in 1945 that "push-button warfare" was just around the corner. These people were closer to being right than they knew, and much closer to being wrong than they thought. By the end of World War II we had entered the era of push-button warfare, but what the writers and speakers of 1945 and subsequent years have overlooked was the fact that, although the necessity for push-button warfare had arrived, the means for meeting the necessity had not. The tempo of modern warfare has increased to such a pitch that when the attack comes we no longer have time even to think; we must base our defense on versatile automatic high-speed devices which we have endowed with intelligence and power to act for us to destroy the attackers or strike a sudden and paralyzing counter-

blow. Properly interpreted, the phrase "era of push-button warfare" becomes a stirring challenge to hard work on the outermost frontiers of scientific knowledge and technical skill, and loses entirely its smug and comforting implications.

The guided missile is one answer to the challenge of the push-button war. It is a modern answer to the problem which has confronted designers of weapons throughout the years: How can we increase range, speed, maneuverability, accuracy, and damage potential without losing reliability or incurring too great a cost? Let us consider one problem in this field—the problem of defending task forces, convoys, or cities against attack by aircraft. Let us assume the attack is made by a large number of high-speed bombers carrying guided bombs so that they are effective at ranges of 5 to 10 miles. The bombers must be engaged and destroyed at ranges well beyond this—a job which cannot be done by conventional guns, even with the best fire-control systems, because of the enormous muzzle velocities and attendant engineering difficulties introduced. After the shell leaves the muzzle of the gun its course in time and space is irrevocably settled—no corrections based on later and better knowledge of the target position are possible. When the time of flight of the projectile becomes large these corrections are of utmost importance if any kind of accuracy is to be attained. This is particularly true in the engagement of nearby targets that move rapidly (such as planes), or distant stationary ones. As the time taken for the projectile to reach the target increases, unpredictable events will occur during the flight time that completely nullify the initial accuracy, for example, evasion by moving targets or action of unknown winds. Consider a bomber flying at 600 mph or 880 fps being attacked by gunfire in which the average time of flight of the shell is 15 seconds. It is easily seen that between the time the shell leaves the gun and arrives at the predicted point of aim, the bomber travels 13,200 ft or $2^1/_2$ miles. In order to reach the point of aim the bomber must fly with an accuracy of one part in 10,000 or, if he is careless and his course deviates by as little as half a degree, he will miss the shell by 106 feet.

On the other hand, the speed of the incoming bombers and the shortness of the time available for effective engagements requires that the speeds and maneuverabilities of the fighter aircraft be increased beyond the present level. The consequent increased fuel consumption will cut down the fighters' idling or cruising time to a low value, reducing the possibility of air patrols. Furthermore, the pilots must rely increasingly on ground-controlled automatic devices to enable them to find and attack their targets—the logical conclusion being a complete replacement of the pilot by mechanical devices. The antiaircraft guided-missile system, therefore, combines qualities of guns and fighter aircraft. It is launched to high speed which it maintains by a propulsion system; during its flight it is continually supplied with information concerning its position with respect to the target;

and it has equipment to make effective use of this information. In order to eliminate delays due to human-reaction times, the whole process of supply and use of intelligence must be made automatic and as rapid as modern electronic servomechanisms and aerodynamic control permit.

In offense, the automatically-guided supersonic missile possesses in virtue of its speed an invulnerability far exceeding that of piloted aircraft. If, in addition, such a missile can be launched deep in friendly territory, its chances of delivering of destructive pay loads far exceed those of any other weapons we know. Among other advantages, this missile makes it unnecessary for us to risk highly trained men in pressing home an effective attack far in enemy-occupied territory.

Essentials of a Guided-Missile System

The challenge of the era of push-button warfare was taken up by the Armed Services of this country. Great advances have been made in the last three years. The road ahead is much more clearly discernible, but we are still a long way from even the first goal.

Let us look for a moment at the features or major components which are essential to any guided-missile system. These major components may be divided into two classes: (*a*) Those that are missile-borne; and (*b*) those that are installed on the launching platform which may be the ground, ship, or large airplane. In Tables 1 and 2 these major components are listed together with a brief description of their functions, and under "implementation" the technical means for supplying or developing them.

TABLE 1 MAJOR MISSILE-BORNE COMPONENTS OF TYPICAL GUIDED-MISSILE SYSTEM

	Major components	*Purpose or function*	*Implementation*
1	War head and fuse	Pay load whose effectiveness characterizes value of whole system	Explosives — fragmentation —other destructive agents
2	Propulsive system	Maintain speed and maneuverability	Jet engines
3	Booster system auxiliary propulsion	Attain high speed with simple platform installations	Rockets
4	System for receiving and interpreting intelligence	Guidance and discrimination—supply information about relative positions of missile and target	Radar or other electromagnetic wave systems. Antennae receivers and components (electronics)
5	Power systems	Implementation of intelligence control	Servomechanisms, electric and hydraulic power
6	Aerodynamic surfaces	Stability and steering, lift implementation of control	Movable wings, fins, or ailerons
7	Airframe	Integration of whole assembly	Structures and material giving optimum strength-to-weight ratios

TABLE 2 PLATFORM-MOUNTED COMPONENTS FOR GUIDED-MISSILE SYSTEMS

	Major components	*Purpose or function*	*Implementation*
1	Launchers and handling equipment	Provide initial direction of missile-readiness	Handling and test equipment. Directable rails for supporting missile, catapults
2	Systems for developing and transmitting intelligence	Guidance	Tracking radars, computers, and coded signals
3	Target acquisition equipment	Determination of positions and motion of target	Means for transferring from search to tracking radars
4	Early warning and identification equipment	Readiness and timely instructions to launch in proper direction	Search radars

It should be emphasized that the attainment of high speed and maneuverability is a most important objective in the development of guided missiles. Hence the total weight must be kept as low as possible, and heroic measures are required to keep the weights of all auxiliary components and structures small enough to leave an allowance for a heavy and effective war head.

Development Organizations

At this point I should like to pause in the discussion of technical aspects of guided missiles to give a short account of one of the organizations engaged in their development, namely, the Applied Physics Laboratory of The Johns Hopkins University, Baltimore, Md. This institution started as a small group of scientists which was organized under the National Defense Research Committee in 1940 to assist the Navy in one of its most important problems, the defense of ships against aircraft. Under the leadership of M. A. Tuve this group worked at the Department of Terrestrial Magnetism of the Carnegie Institution of Washington and undertook as the first technical solution, the Navy's problem, the development of radio proximity (later called VT) fuses for antiaircraft guns. This development sought to endow a shell with a rudimentary intelligence to decide for itself upon closer inspection of the target the proper time to explode. You will recall that in time fuses this decision had to be made before the shell was fired. The group grew rapidly to more than 1000 and enlisted the co-operation of industrial and academic organizations throughout the country. In March, 1942, the group moved to the present Applied Physics Laboratory in Silver Spring, Md., and at the same time The Johns Hopkins University undertook responsibility for the operations of the Laboratory under contract with the Office of Scientific Research and Development. After the first radio proximity fuse had been well launched into production, the Applied Physics Laboratory started on another problem in the same field, namely, the development of compact gunfire-control systems to make

the auxiliary batteries on warships more flexible and effective in defense against multiple attacks. The gunfire-control systems, Mark 57 and Mark 61, are the results of these developments.

In July, 1944, it became evident that radically new equipment would be required to enable the defenders of ships to cope with the menace of the suicide plane and its unholy mother or, more generally, with the bomber capable of attacking effectively at long ranges by guided bombs or missiles. The Bureau of Ordnance, U. S. Navy, asked the Laboratory to undertake a study of this problem, and the technical answer given was that guided missiles should be developed for antiaircraft defense of ships. In December, 1944, the contract covering operations of the Applied Physics Laboratory was transferred from the Office of Scientific Research and Development to the U. S. Navy, and in January, 1945, the Bureau of Ordnance assigned to the Laboratory a broad task covering research and development leading to guided missiles. The code name "Bumblebee" was given to the program. The formulation of detailed technical objectives under this task was delegated to the laboratory which was empowered to construct the necessary facilities and enlist the co-operation of academic and industrial organizations in carrying out the task. Approximately 14 industrial organizations and ten universities have joined with the Applied Physics Laboratory in this enterprise, making a family of Associate Contractors whose varied knowledge, skills, and facilities enabled it collectively to handle all phases of the research, development, and engineering required to bring to realization a modern guided missile. These contractors operate under contracts made directly between them and the U. S. Navy. These contracts, called Section T Series contracts, the name originating in the early OSRD days, provide for the execution of broad tasks in accordance with a program formulated and integrated by the director of the Applied Physics Laboratory. It may be remarked that the Bureau of Ordnance has given wholehearted support to this organization and program, not only by funds but by a genuine interest in developing a mechanism whereby the Armed Services may secure the most effective co-operation by the civilian scientists and engineers in research and application of technical knowledge to the solution of military problems. The system has been working well for approximately four years and offers a method whereby a complex program of broad scope may be carried out with widespread initiative coupled with reasonable integration.

The Applied Physics Laboratory itself is organized to carry out research and development work in all fields associated with the program, namely, propulsion, aerodynamics and control, radar and intelligence, launching, war-head design, composite design, telemetering, and ground and flight testing. One or two other contractors also cover the whole field with emphasis on the engineering and fabrication aspects, but the majority confine their efforts to one or two specific fields. In all the fields of technical ac-

tivity Bumblebee panels have been organized and all interested contractors are represented on them. Since the panels play a large part in the initiation and evaluation of technical programs, they provide instruments whereby a broad base of scientific thought may be established and utilized in the formulation and execution of the program, and whereby all interested contractors may participate in the planning as well as the carrying out of the work. You will get an idea of the magnitude of the operation when I tell you that in 1948 APL and its Associate Contractors issued 100 formal and 430 informal reports.

Specialized Facilities

The original objective of the Bumblebee program was the development of rocket-launched ramjet-propelled supersonic missiles fitted with practical guidance systems. In 1945, ramjets were known only in theory and stability, and control at supersonic speeds lay in unexplored territory. The answers to many other vital questions were also unknown. It was imperative that facilities be constructed for experimental work on ramjets and supersonic aerodynamics. By assembling surplus compressors, an air supply to permit study of supersonic ramjets up to 6 in. diam was established at the Forest Grove station of the Applied Physics Laboratory and work was under way in July, 1945. At the same time an Associate Contractor, the Standard Oil Development Corporation, also built a laboratory for testing 6-in. ramjets, and other contractors, notably the University of Virginia and Experiment Incorporated, assembled apparatus for smaller scale experiments. Indeed the testing of small-scale ramjets began as early as February, 1945, at the University of Virginia. The need for larger facilities was also foreseen and early in 1945 a large idle air supply was found at a blast furnace near Daingerfield in Texas. By co-operation with the Navy, the company owning the blast furnace, and the Consolidated Vultee Aircraft Corporation, what is now known as the Ordnance Aerophysics Laboratory, was established by APL for studying large scale ramjets and supersonic aerodynamics. Ramjet testing began as early as August, 1945, at this facility.

Under contract with the Navy, the Pittsburgh-Des Moines Steel Company built a continuous-flow supersonic wind tunnel capable of operating between Mach numbers 1.25 and 2.5 at convenient intervals. The test section of this tunnel is 19 by 27 inches. Work on the tunnel started in June, 1945, and was essentially completed by November, 1946. The tunnel was in full operation by March, 1947. Charles Looney and his associates at APL had guided a co-operative program that led rapidly to one of the country's most useful supersonic wind tunnels. At present the whole Ordnance Aerophysics Laboratory is operated by the Consolidated Vultee Aircraft Corporation under Section T type contract with the Navy and in

accordance with instructions of the director of the Applied Physics Laboratory supported by the advice of representative panels.

Aerodynamics and Control

Analyses of the kinds of motion missiles will be called upon to execute given preliminary requirements for maneuverability, stability, and airframe strength, etc., which define problems in supersonic aerodynamics, such as: design of air frames; optimization of lift and drag; determination of lift, drag, pitching and rolling moments, including interference phenomena in composite bodies; changes of center of pressure; adjustment of trim in combined motions; and over-all stable response to signals and hinge moments of control surfaces. It will be seen that the aerodynamics of the missile form a portion of the servocontrol loop, and the determinations of the various aerodynamic coefficients, their variation with angle of attack and Mach number supply basic data for design. At first, chief reliance was placed on observations telemetered from freely flying models. With the completion of the supersonic wind tunnel, the procedure now is to make as many measurements as possible in the tunnel and check the results with telemetered observations on missiles or test vehicles in free flight. Many of the preliminary designs have been made solely on the basis of theory with satisfactory results. Unexpected phenomena have been observed, such as the reversal of response to roll-control ailerons because of wing and body interference effects with certain common types of composite bodies, or the effect of wing-body interactions on the total lift, generated.

These effects were discovered experimentally, but they can be accounted for on sound theoretical basis and there is no reason to doubt the adequacy of present supersonic theory, or to believe that fundamental obstacles exist in the way of controlling missiles in supersonic flight.

Transmission, Receipt, and Interpretation of Intelligence

In order that a guided missile may know where it is and whither it is supposed to go, it must receive intelligence from the outside; this intelligence must be expressed in terms of a fixed set of co-ordinates so that it can be interpreted by the missile.

In certain cases the information as to where and whither are given to the missile at the launcher and no further information is imparted after the missile is launched. Such a system is called a preset guidance system—the German V-1 and V-2 missiles are notable examples of the use of such systems. Even in this case the missile carries components such as gyros, gyro compasses, altimeters, velocity meters, and the like which enable it to sense its path. A present system is particularly applicable in such missiles as the V-2 where the main portion of the path lies along a ballistic trajectory, and where the rocket motor and the guidance system serve the

same purpose as a gun barrel many tens of thousands of feet long. By suitable mechanical devices in the V-2, the rocket motor is cut off when the velocity reaches a certain value at the same time its direction of flight is so oriented that a vacuum trajectory will bring it to the target. From then on no corrections from external sources are made. In most cases the missile will receive information throughout its flight and correct its course accordingly. The links between a missile and the outside world are normally through electromagnetic radiation including light long-wave radio, or continuous or pulsed microwave radar. Four general systems are under study.

Command Guidance: This term is applied to line-of-sight radio or radar control where a ground station tracks both the missile and the target and then computes what adjustments should be made to the missile's course to bring the two into collision. Instructions generated from these computations are sent to the missile through the electromagnetic link and its course is altered accordingly. In this system the missile only needs to carry enough mechanism to receive and obey commands from the ground. Its order of intellect is, therefore, quite low. The ground installation, however, must have considerable rapid computation ability.

Way Following Guidance: This is also essentially a line-of-sight guidance system. An artificial path from the launcher to the target is marked out in space, a radar beam tracking the target being such a path, and the missile is provided with a mechanism for sensing continuously its position with respect to the axis of the path, knowng when it deviates and correcting its course accordingly. The "beam rider" is the best investigated example of this type of guidance. A beam-rider missile is much more complicated; it has a higher order of intellect than a command-guided missile. On the other hand, many missiles may ride the same beam and the ground installations are comparatively simple. Beam-rider guidance imposes more stringent limitations on the missile's propulsion system than does command guidance.

Homing Guidance: This calls for a still higher order of intelligence in the missile. In this system the missile receives a directional signal directly from the target and adjusts its course to bring about a collision. Thus the missile recognizes its quarry and goes after it. Such systems offer the possibility of great accuracy in the terminal phases of the missile's course but, as you can see, they suffer from range restrictions and are usually used in conjunction with other systems for initial and midcourse guidance.

Navigation: By observing its position in a natural frame of reference such as that given by the stars and the direction of the vertical, or in an artificial one such as that set up by an electromagnetic network in Loran, a missile may adjust its motion so as to follow a predetermined course from the launching point to the target. It is no easy matter to accomplish this reliably and accurately, but the long-range missile will probably use

such methods of guidance. A special case of this type of system is the dead reckoning one in which a frame of reference is defined in inertial space by gyros and the motion of the missile followed by accelerometers. Here again a knowledge of the vertical is required.

Stability and Response to Intelligence—Simulators

If you analyze carefully your reactions as you drive an automobile, airplane, or even a bicycle at high speed, you will find that instinctively you make a number of adjustments not only to correct for deviations from the desired course but also to anticipate the results of these corrections to see that the vehicle does not overshoot too much. The importance of anticipation in steering is well known to anyone who has tried to manage a boat. Stability of flight—the avoidance of unduly sharp turns or large oscillations about the correct path—is a problem of prime importance in the control of guided missiles. After a missile receives intelligence, a complex chain of events is set up culminating in the response of the missile to movement of the aerodynamic control surfaces. It is essential that this chain of circumstances leads to convergent and not divergent oscillations about the path, and the prediction of this convergence involves complex mathematical computations. In order to study flight stability, special purpose computing machines called flight simulators have been devised. Into these machines is fed information about the aerodynamic coefficients, the parameters of electrical circuits and servomechanism. The result is a curve or series of curves which show the type of trajectory the missile will fly. It is also possible to use the actual servomechanisms and intelligence components of the missile so that the control-surface deflections are generated precisely as in an actual flight. The simulator then simply computes the aerodynamic link in the flight, that is, the simulator computes the change in received intelligence that would result from the control surface deflections obtained. With such a simulator a man can do the equivalent of flying 50 to 100 missiles in a morning and determine the adjustments to be made to give the highest probability of success in an actual flight test.

From the technical point of view the guidance problems are those concerned with: more powerful and more precise radars, and other means of electromagnetic propagation; increase in the sensitivity and reliability of electronic components; development of compact and light power supplies; increase in the reliability of free and rate gyros, star trackers, accelerometers and other means of sensing motion in a frame of co-ordinates; recognition and discrimination of targets; the development of responsive stable servomechanisms and their coupling to the surroundings by control surfaces, rockets or other means.

Propulsion Systems for Supersonic Missiles

In general a supersonic guided missile requires two propulsion systems:

(a) The auxiliary propulsion system or booster to accelerate it to approximately its design speed; and (b) the main propulsion system to maintain it at this speed against the retarding action of gravity or of aerodynamic drag forces throughout its flight and maneuvers. It should be emphasized that both these propulsion systems are required to gain and maintain the high speeds which are characteristic of the supersonic guided missile, and which give it flexibility and surprise in attack and invulnerability in defense.

In this respect the subsonic V-1, which was boosted to velocity by a large hydrogen-peroxide-fueled gun and subsequently propelled by a pulse jet, is more typical than the V-2 which is propelled from rest by its main propulsion system. I have already noted that the V-2 flies a ballistic trajectory, does not maneuver, and has no midcourse or terminal guidance. It can, therefore, afford to carry a large power plant with it and the speed of launching is not critical.

Generally speaking the thrust required to give a supersonic missile the appropriate acceleration is approximately ten to twenty times the thrust required to maintain its cruising speed. It may be mentioned that accelerations of the order of 30 g are sufficient to bring the missile to speed in a reasonable time. Few lightweight power plants are versatile enough to give this range of thrust, and it pays to employ an auxiliary one which may be dropped when most of the acceleration is over, thereby lightening the weight and enhancing the maneuverability of the missile.

General requirements for the propulsion systems of guided missiles are: (1) High thrust per unit frontal area; (2) high thrust per unit weight; (3) reasonable fuel economy; and (4) expendability, simplicity, and high reliability for short times. Supersonic flight makes stringent demands in items 1 and 2. As you know, the aerodynamic drag on a body is given by

$$D = {}^1/_2\, \rho\, v^2 C_D A \qquad [1]$$

It is proportional to the frontal area A, the drag coefficient C_D and the density ρ and to the square of the velocity v. You will also recall that C_D is greater at supersonic than at subsonic speeds. Thus, the thrust to overcome drag at supersonic speeds goes up somewhat more rapidly than the square of the velocity for a given frontal area. The horsepower which is the product of thrust and velocity, therefore, rises more rapidly than the cube of the speed. Thus the horsepower required to cruise at 1600 mph (2340 fps) is 64 times that required to cruise at 400 mph for a vehicle of the same frontal area and configuration. Furthermore, while lift-to-drag ratios exceeding 20 to 1 are easily realized in subsonic vehicles, a reasonably good lift-to-drag ratio in supersonic flight is 4 to 1. Heavy power plants thus impose a heavy penalty through the increased drag associated with the extra lift required to support them.

It is well recognized that jet engines are the only ones that fulfill the

requirements we have laid down, and four types of jet engines are now available for missile propulsion, namely: turbojets; pulse jets; ramjets; and rockets. It will be seen that the rocket and ramjet are in a class by themselves as regards horsepower per unit area, especially at high velocities. On the other hand, the specific fuel consumptions of rockets are much higher than those of any other engine at all speeds. The specific fuel consumption of ramjets is much higher than the other air-breathing engines at low speeds, but at high speeds it drops to a value comparable with that of even reciprocating engines on a thrust-horsepower basis.

TABLE 3 SUMMARY OF CHARACTERISTICS OF JET ENGINES

Engine	*Typical operating range Mach number*	*Characteristics*	*Advantages*	*Disadvantages*
Pulse Jet	0 –0.6	Air breathing; ram pressure and vanes; intermittent operation 2.8 lb fuel/hr/lb thrust	Thrust at zero velocity; cheapness and ease of fabrication; cheap fuels	Questionable operation at high altitude and supersonic speeds; severe vibration
Turbojet	0 –0.8	Air breathing; mechanical compressor and turbine; continuous operation; 1.2 lb fuel/hr/lb thrust at M = 0.8	Thrust at zero velocity; reasonable specific fuel consumption; cheap fuels	Mechanical complexity and cost
Ramjet	1.5 –3.5	Air breathing; ram pressure and diffuser; continuous operation; 2 lb fuel/hr/lb thrust at M = 3.	No moving mechanical parts; high thrust per unit area and weight; cheap fuels; reasonable specific fuel consumption at high speeds	No thrust at zero velocity; design critical to flight speed
Rocket (liquid fuel)	....	Does not require air; continuous operation; 1.5 lb fuel/hr/lb thrust (alcohol-oxygen)	Thrust independent of velocity; improves with altitude; adaptable to compact packaging	Fuel and oxidizers expensive and hard to handle; high specific fuel consumption; severe cooling problems
Rocket (solid fuel)	....	Does not require air; continuous operation	Thrust independent of velocity; high thrust for short duration; great simplicity of construction and operation	Short duration; high specific fuel consumption; severe thermal problems

From the foregoing discussion it will be seen that in the present state of the art three power plants are available for the propulsion of *supersonic* missiles, namely: solid fuel rockets; liquid fuel rockets; and ramjets. The expendable supersonic turbojet could be a competitor of the ramjet, but radical advances in its design and construction toward simplification on the one hand, and high energy release per unit volume on the other, are required before this competition becomes serious.

(*To be continued in the December issue*)

American Rocket Society News

1949 ARS National Convention to Be Held at Hotel Statler, New York, N. Y., Nov. 28-Dec. 2

THE fourth annual National Convention of the AMERICAN ROCKET SOCIETY will be held in conjunction with the 1949 Annual Meeting of The American Society of Mechanical Engineers, Hotel Statler, New York, N. Y., Nov. 28–Dec. 2, 1949.

Annual Convention program follows:

THURSDAY, DECEMBER 1

Session 2

9:30 a.m.

Fundamental Problems in Rocket Research, by Martin Summerfield, editor, Aeronautics Publication Program, Princeton University, Princeton, N. J.

The Use of Strategic Materials on Jet and Rocket Applications, by Marvin C. Demler, colonel, United States Air Force, Arlington, Va.

Session 3

2:30 p.m.

Instruction and Research in Jet Propulsion at the Guggenheim Jet Propulsion Center at California Institute of Technology, by H. S. Tsien, director, Florence and Daniel Guggenheim Jet Propulsion Center at the California Institute of Technology, Pasadena, Calif.

Research in Jet Propulsion at the Florence and Daniel Guggenheim Jet Propulsion Center at Princeton University, by Luigi Crocco, Robert H. Goddard Professor, Princeton University, Princeton, N. J.

Performance Parameters for Several Applications of Rocket Power Plants, by G. P. Sutton, supervisor, propulsion development aerophysics laboratory, North American Aviation, Inc., Downey, Calif.

Honors Night and Dinner

6:30 p.m.

Presiding: G. Edward Pendray, president, Pendray and Leibert, Inc., New York, N. Y.

Subject: The Complexities of Peace

Speaker: Dan A. Kimball, Under Secretary of Navy, Washington, D. C.

Awards: Goddard Award to Admiral C. M. Bolster, U.S.N.; Hickman Award to James A. Van Allen, director, Applied Physics Laboratory, Johns Hopkins University, Baltimore, Md.; ARS Student Award to Leon Cooper, College of the City of New York.

FRIDAY, DECEMBER 2

Session 4

9:30 a.m.

Hydrogen Peroxide as a Rocket Fuel, by Noah S. Davis, Jr., special projects department, Ralph Bloom, Jr., and Samuel D. Levine, project supervisors, Buffalo Electro-Chemical Company, Buffalo, N. Y.

Nitrogen Textroxide as a Rocket Fuel, by D. H. Ross, assistant manager, products development department, The Solvay Process Division, New York, N. Y.

The Handling of Liquid Oxygen, by G. E. Simpson, division superintendent of distribution, Linde Air Products, Inc., New York, N. Y.

In addition to these, the AMERICAN ROCKET SOCIETY will co-sponsor a technical session and a luncheon with several professional divisions of the ASME.

MONDAY, NOVEMBER 28

8:15 p.m.

Session 1

Co-sponsored with the ASME Industrial Instruments and Regulators and the Aviation Divisions

Automatic Temperature Control for Electrically Heated Windshields, by Harry R. Karp, engineer, engineering department, eclipse-pioneer division, Bendix Aviation Corporation, Teterboro, N. J.

An Automatic Machine for Analyzing Telemetered Missile Data (Hermograph), by B. S. Benson, engineer, missiles design group, Douglas Aircraft Company, Inc., Santa Monica, Calif.

TUESDAY, NOVEMBER 29

Luncheon

12:15 p.m.

Co-sponsored with the ASME Gas Turbine Power, Aviation, and Power Divisions

Presiding: J. K. Salisbury, thermal power systems division, General Electric Co., Schenectady, N. Y.

Toastmaster: Alex D. Bailey, Vice-President, Commonwealth Edison Co., Chicago, Ill.

Speaker: Hugh L. Dryden, director of research, The National Advisory Committee for Aeronautics, Washington, D. C.

Subject: Research Airplanes

Members of the American Rocket Society are entitled to attend all sessions at the ASME annual meeting whether they are American Rocket Society sessions, co-sponsored sessions, or ASME sessions.

Every member of the ARS will be interested in the program planned for the ARS National Convention this year. Keep the week of November 28 to December 2 open. Do not fail to attend the sessions and dinners.

ARS Section Activities

THE activity of the National Organization at this time is centered upon the fourth Annual Convention to be held at the Hotel Statler, New York, N. Y., from Nov. 28 to Dec. 2, 1949. The program given in this issue of the JOURNAL speaks for itself.

New York Section

The first meeting of the New York Section was held on Sept. 16, 1949, at the Engineering Societies Building, New York, N. Y. William Smith, rocket installation engineer, Bell Aircraft Corporation, Buffalo, N. Y., addressed a meeting of approximately 75 members on "Rocket Engine Installation Problems." The talk was well received.

The New York Section joined the Institute of Navigation on Oct. 18, 1949, in a dinner meeting held at Miller's Restaurant at which Henry Blackstone, Servo Corporation of America, spoke on "Navigation in the Upper Atmosphere."

The ballots for the election of three members of the Board of Directors have been sent to all active members in the New York Section. Balloting must be completed by Dec. 1, 1949.

The New York Section is arranging a full schedule for this season including the showing of foreign movies and talks by outstanding speakers.

Southern California Section

The fall program of the Southern California Section opened on Sept. 22, 1949, with the presentation of a paper by E. L. Wilson, research engineer, Jet Propulsion Laboratory, GALCIT. The meeting was a joint one with the Heat Transfer Division of the ASME, and was held at the Arms Laboratory, California Institute of Technology.

On Oct. 19 another joint meeting with the ASME was held at UCLA. The speaker was Colonel Harold R. Turner, deputy commander, Joint Long Range Proving Ground. Colonel Turner, formerly commanding officer of White Sands Proving Grounds, spoke on the new Joint Projects at Cocoa, Fla., which he heads.

On Oct. 20 the first security classified session of the Southern California Section was held at the plant of the Aerojet Corporation, Azusa, Calif. A plant-inspection tour was made. Several confidential papers dealing with operational rocket problems were presented. Admission to this session was limited to persons with proper security clearance.

Dan A. Kimball to Speak at 1949 ARS National Convention

DAN A. KIMBALL who will be the main speaker at the ARS Annual Banquet to be held in the Hotel Statler, New York, N. Y., Dec. 1, 1949, was born in St. Louis, Mo., on March 1, 1896, but has been engaged in business in California for the last 34 years. For the past several years he and Mrs. Kimball, the former Dorothy Ames, have made their home in Los Angeles at 2221 West Live Oak Drive.

Mr. Kimball was an Army Air Corps aviator in World War 1. He trained in the same group as General James H. Doolittle, received his commission as second lieutenant in the Army Air Corps on March 1, 1918, and mustered out at the end of the war as a first lieutenant. Shortly after his separation he became associated with the General Tire and Rubber Company of Akron, Ohio.

For many years Mr. Kimball managed the business of this company in eleven western states. In 1944 he was placed in immediate charge of the Aerojet Engineering Corporation, at Azusa, Calif., a subsidiary of General Tire and Rubber Company. Shortly thereafter he was made a vice-president and director of the parent company.

As head of the Aerojet Engineering Corporation, Mr. Kimball has taken a leading part in the development of rockets and other unorthodox means of propulsion. Among the products of the company is the equipment for jet-assisted take-offs for aircraft.

Mr. Kimball was nominated by the President of the United States as Secretary of the Navy for Air on February 11, 1949, and confirmed by the United States Senate on February 25, 1949.

On May 13, 1949, Mr. Kimball was appointed by the President of the United States as Under Secretary of the Navy, and the appointment was confirmed by the United States Senate on May 19, 1949.